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LEVEL

PRELIMINARY AIRWORTHINESS EVALUATION

OH-58C HELICOPTER

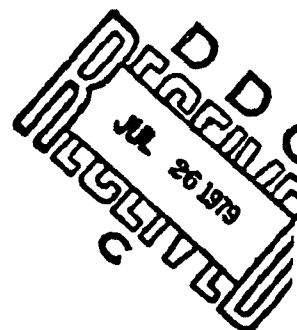
FINAL REPORT

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UNITED STATES ARMY AVIATION ENGINEERING FLIGHT ACTIVITY
EDWARDS AIR FORCE BASE, CALIFORNIA 93523

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20. Abstract

- > maximum gross weight (3200 pounds) and aft center of gravity (fuselage station 112) conditions. The possibility of inadvertent activation of the night vision goggle switch located on the pilot cyclic control grip was identified as a deficiency. Activation of this switch during day flight rendered the warning and caution lights unreadable. The most significant difference in flying qualities between the OH-58C and the OH-58A was a pitch-up tendency at cruise airspeed and aft center of gravity, which is a shortcoming. Insufficient left directional control in right sideward flight, excessive yaw oscillations in left sideward flight between 15 and 30 knots true airspeed, and excessive pitch and yaw oscillations in rearward flight are also shortcomings. Eight additional shortcomings were noted.

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DEPARTMENT OF THE ARMY
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DRDAV-EQ

APR 20 1979

SUBJECT: Director of Development and Engineering's Position on the
Final Report of USAAEFA Project No. 76-11-1, Preliminary
Airworthiness Evaluation, OH-58C Helicopter, March 1978

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1. The Directorate for Development and Engineering's position on the subject report is provided herein. Paragraph numbers from the subject report are provided for reference.

a. Paragraph 39 and 43 - To prevent accidental transfer to the night vision goggle (NVG) low intensity light levels, a NVG "ENABLE" switch has been installed in the cockpit area of all OH-58C's. Transfer to NVG light levels shall only be possible when the NVG "ENABLE" switch is activated. Deactivation of the NVG "ENABLE" switch shall be accomplished either manually or automatically whenever electrical power is removed from the system.

b. Paragraph 40a - Bell Helicopter Textron (BHT) is investigating the use of leading edge slats on the horizontal stabilizer and/or other aerodynamic methods to solve this problem.

c. Paragraph 40b, c & d - Revised hover and wind azimuth charts will be added to the Operator's Manual to reflect sideward and rearward flight limitations. An improved tail rotor system development and qualification effort will soon be on contract with hardware kits available approximately FY 83.

d. Paragraph 40e - Similar trim control displacement bands are found on the OH-58A and have not been reported as objectionable.

e. Paragraph 40f - It is possible that this shortcoming may be corrected while correcting the pitch-up tendency at cruise airspeed.

f. Paragraph 40g - The transponder (AN/APX-100) and the communication security set (TSEC/KY-28) were interchanged which helped alleviate the problem. There is no immediate solution short of redesigning the readout windows and relocating the transponder in the instrument panel.

g. Paragraph 40h - This is similar to the OH-58A and has not been reported as objectionable.

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APR 20 1979

SUBJECT: Director of Development and Engineering's Position on the
Final Report of USAAFPA Project No. 76-11-1, Preliminary
Airworthiness Evaluation, OH-58C Helicopter, March 1978

h. Paragraph 401, j & k - A production change was incorporated in 1969 to install two diodes to correct this problem. However, the OH-58A's already fielded were not retrofitted. All OH-58A's not having this electrical modification incorporated will be so modified. This should correct these electromagnetic interference problems.

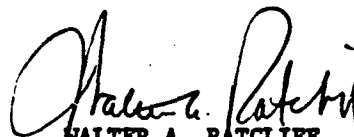
i. Paragraph 401 - The ON-OFF labels were inadvertently left off the test aircraft. All production aircraft have the proper labels installed.

j. Paragraph 44 - Many of the shortcomings listed are found on the OH-58A and have not been reported as objectionable. Money and complexity of design change do not warrant correction.

k. Paragraph 45 - Appropriate cautions and notes have already been incorporated in the Operator's Manual.

2. In summary, from an airworthiness point of view the OH-58C is considered an improved helicopter over the OH-58A, primarily as a result of the incorporation of the Allison T63-A-720 engine and improved instrument lighting providing night vision goggle compatibility, which are significant safety inputs in the design. Operational tests have raised questions regarding the impact of the flat plate canopy design on the suitability of the OH-58C as a scout helicopter due to reduction in overall external visibility; however, these problems are addressed in other test reports.

FOR THE COMMANDER:



WALTER A. RATCLIFF
Colonel, GS
Director of Development
and Engineering

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INTRODUCTION

BACKGROUND

1. The United States Army Aviation Systems Command (AVSCOM*) awarded a contract to Bell Helicopter Textron (BHT) in June 1976 to modify three OH-58A helicopters to a prototype OH-58C helicopter configuration. Primary objectives of the modification were to provide improved hover and vertical climb performance and reduced ballistic vulnerability. In September 1976, AVRADCOM directed the United States Army Aviation Engineering Flight Activity (USAAEFA) to prepare a test plan for an Army Preliminary Evaluation (APE**) of the prototype OH-58C helicopter (ref 1, app A). The original test scope was revised by a subsequent message (ref 2) and a test plan was prepared in December 1977 (ref 3). Flight testing by USAAEFA commenced 21 February 1978.

TEST OBJECTIVES

2. The objectives of the OH-58C PAE were as follows:
 - a. Provide a limited assessment of the handling qualities of the OH-58C.
 - b. Provide quantitative and qualitative engineering flight test data for comparison with data previously obtained for the OH-58A.
 - c. Detect and allow for early correction of any deficiencies or shortcomings.

DESCRIPTION

3. The OH-58C helicopter is a derivative of the OH-58A built by BHT. The OH-58C has a single two-bladed, semirigid, seesaw-type main rotor and a single two-bladed, semirigid, delta-hinged tail rotor. The main and tail rotor configurations are unchanged from the OH-58A model. The design gross weight of the OH-58C is 3200 pounds. The aircraft is powered by an Allison T63-A-720 free turbine engine with an uninstalled standard-day, sea-level intermediate rated power (30 minutes) of 420 shaft horsepower (shp). The helicopter transmission has a 5-minute rating of 317 shp and a continuous rating of 270 shp. The cockpit provides side-by-side seating for a crew of two (pilot and copilot/observer). Dual mechanical flight controls are provided. The cyclic and collective control systems are hydraulically boosted and are essentially unchanged from the OH-58A systems. Redundant unboosted directional control systems are installed. The primary system

*Since redesignated United States Army Aviation Research and Development Command (AVRADCOM).

**Since redesignated Preliminary Airworthiness Evaluation (PAE).

consists of push-pull tubes and the backup system consists of a push-pull cable. In addition to the new engine, other major changes incorporated in the OH-58C are flat-plate cockpit canopy, low reflective (LR) fuselage paint, infrared suppressive engine cowlings and exhaust stacks, tail rotor drive shaft cover, and redesigned cockpit instrument panel incorporating new indicators and avionics. A further description of the OH-58C helicopter is presented in appendix B.

4. The primary test helicopter, SN 68-16850 (photos A and B), was a prototype and was similar in external configuration to the production aircraft. The test helicopter incorporated all external OH-58C modifications except for the LR fuselage paint. A flight test boom extended forward of the aircraft and was attached beneath the nose in the area where the landing light is normally installed. An instrumentation slip ring assembly was installed on top of the main rotor mast. The cockpit instrument panel was nonstandard. Numerous test indicators and instrumentation controls were installed in place of standard instruments and avionics. The instrumentation signal conditioning equipment and tape recorder were installed in the passenger/cargo compartment of the aircraft.

5. A second noninstrumented prototype helicopter, SN 69-16214, was also used. This aircraft incorporated all OH-58C modifications and was representative of the production aircraft. Qualitative evaluations of the cockpit arrangement, systems operation, checklist procedures, and night flying were conducted in this aircraft.

TEST SCOPE

6. The PAE of the OH-58C was conducted at the Bell Helicopter Flight Research Center, Arlington, Texas (elevation 630 feet), from 21 February 1978 through 4 March 1978. A total of 18 flights were conducted for a total of 19.8 flight hours. Productive test time was 14.7 hours. BHT supplied, calibrated, and maintained the test instrumentation and performed aircraft maintenance during the test. Testing was conducted in accordance with the test plan. General test conditions are shown in table 1. Flight restrictions and limitations observed during the PAE are contained in the operator's manual (ref 4, app A) and the airworthiness release (ref 5). Test results were analyzed with respect to the OH-58C prototype detail specification (ref 6) and military specification MIL-H-8501A (ref 7), and compared with previous USAAEFA test results (refs 8 through 11).

TEST METHODOLOGY

7. Established flight test techniques and procedures were used (ref 12, app A). The test methods and data analysis methods are briefly described in appendix D of this report. A Handling Qualities Rating Scale (HQRS) (fig. 1, app D) was used to augment pilot comments relative to handling qualities. The test data were obtained from test instrumentation displayed on the instrument panel and recorded on magnetic tape on board the aircraft. A detailed listing of the test instrumentation is contained in appendix C.

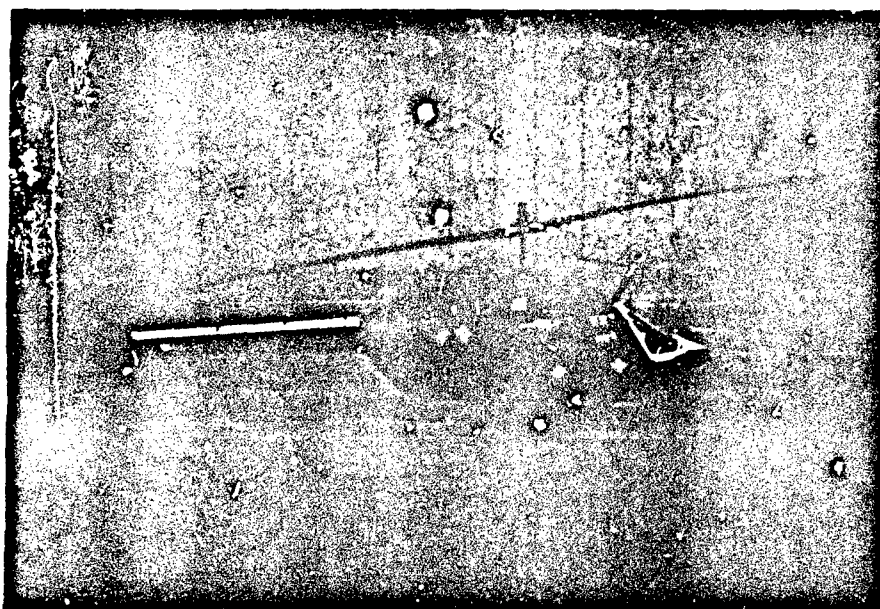


Photo A. Test Helicopter Right-Side View.

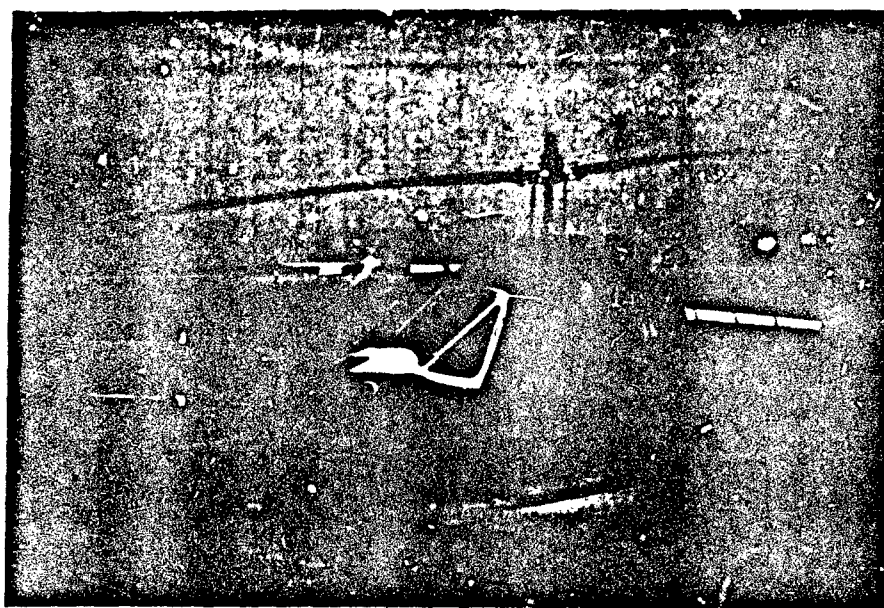


Photo B. Test Helicopter Left Quarter View.

Table 1. Flight Test Conditions.

Test	Average Gross Weight (lb)	Average Center-Gravity Location ¹ (FS)	Average Density Altitude (ft)	Average Rotor Speed (rpm)	Trimmed Calibrated Airspeed (kt)	Remarks
Control positions in trimmed forward flight	3180	107.2 (fwd)	5120	354	50	Climb ²
					33 to 116	Level flight and dive ³
					46	Autorotation
					50	Climb ²
Static longitudinal stability	3140	112.3 (aft)	5080	352	29 to 116	Level flight and dive
					49	Autorotation
					51, 96	Level flight
					75	Partial power descent ⁴ and autorotation
Static lateral-directional stability	3150	112.3 (aft)	5570	353	51, 97	Level flight
Maneuvering stability	3140	112.3 (aft)	5290	353	49, 96	Steady-state turns
		111.2 (aft)		357	86, 96	
Dynamic stability	3130	112.1 (aft)	7870	354	51, 94	Level flight
Controllability	3130	112.1 (aft)	7660	354	94	Level flight
Low-speed flight	3140	107.2 (fwd)	-1560	358	Note ⁵	10-foot skid height
Simulated engine failure	3120	107.1 (fwd)	5130	354	49, 96	Level flight
					53	Climb ²

¹Average lateral cg: BL 0.2 right.²Climb at takeoff power.³Dive at maximum continuous power.⁴Partial power descent at 500 feet per minute.⁵Airspeed ranges: Zero to 38 knots true airspeed (KTAS) forward; zero to 32 KTAS left sideward, zero to 33 KTAS right sideward; zero to 30 KTAS rearward.

RESULTS AND DISCUSSION

GENERAL

8. Primary emphasis during the PAE flight testing was placed on evaluating the aircraft at the maximum gross weight limit (3200 pounds) and aft cg condition (FS 112.2). The possibility of inadvertent activation of the night vision goggle switch located on the pilot cyclic control grip is a deficiency. Activation of this switch during day flight rendered the warning and caution lights unreadable. The most significant difference in flying qualities between the OH-58C and the OH-58A was a pitch-up tendency at cruise airspeed and aft cg, which is a shortcoming. This pitch-up tendency was objectionable during maneuvering flight and required increased pilot compensation to control load factors following abrupt aft longitudinal cyclic control inputs. Three shortcomings identified during sideward and rearward flight are insufficient left directional control in right sideward flight, excessive yaw oscillations in left sideward flight between 15 and 30 KTAS, and excessive pitch and yaw oscillations in rearward flight. These shortcomings increased pilot workload during low-air-speed maneuvering flight and during hover in left crosswind or tail wind conditions. Eight additional shortcomings were identified.

HANDLING QUALITIES

Control System Characteristics

9. The mechanical characteristics of the OH-58C flight control system were measured on the ground with the rotor and engine stopped. All adjustable control friction devices were OFF and force trim was ON. Hydraulic and electrical power were provided by an external source. The cyclic and collective controls are hydraulically boosted.

10. The limits of longitudinal and lateral cyclic control travel are presented in figure 1, appendix E. The variation of control position with applied control force for the longitudinal and lateral controls is presented in figures 2 and 3. The longitudinal and lateral cyclic control force gradients were positive and essentially linear with no discontinuities. A summary of the cyclic control system mechanical characteristics is presented in table 2. Longitudinal and lateral centering characteristics were positive but did not return the control precisely to the original position. This resulted in 1.7-inch longitudinal and 1.2-inch lateral trim control displacement bands. When the cyclic control was displaced from trim, it would not return to its original trim position unaided. Similar trim control displacement bands were documented on the OH-58A but were not reported as objectionable (ref 8, app A). In flight, the large trim control displacement bands resulted in increased pilot compensation to maintain desired attitude and airspeed (HQRS 4). The large longitudinal and lateral cyclic trim control displacement bands are a shortcoming.

11. The directional control breakout force (including friction) for the primary system was 6.2 pounds right and 6.0 pounds left. These forces were measured from a neutral pedal position. For the backup directional control system (measured with the primary system disengaged), the breakout force (including friction) was

4.5 pounds in both directions. Total directional control travel for the primary and backup systems was 6.1 inches. The directional control system did not incorporate a force trim mechanism; therefore, no force gradient or control centering existed. No significant differences were observed in the directional control of the aircraft in flight when the primary directional control system was disengaged and the backup control system was used. Negligible control position free play (less than 0.1 inch) was noted in the directional control system.

Table 2. Longitudinal and Lateral Control System Mechanical Characteristics Summary¹.

Parameter	Control	
	Longitudinal	Lateral
Control travel (in.)	11.9	9.5
Breakout force including friction (lb)	1.1 fwd	1.0 left
	1.5 aft	1.4 right
Average force gradient near trim (lb/in.)	0.9 fwd	0.5 left
	1.1 aft	0.5 right
Average friction band near trim (lb)	2.0 fwd	1.75 left
	2.0 aft	1.75 right
Trim control displacement band (in.)	1.7	1.2
Centering	Positive fwd and aft	Positive left and right
Control position free play (in.)	<0.1	<0.1
Stick jump	Negligible	Negligible

¹Hydraulic boost and force trim ON, adjustable control friction OFF.

12. The collective control breakout force (including friction) was approximately 3 pounds when pulling up from a full-down position and approximately 2.5 pounds when pushing down from a full-up position. Total collective control travel was 10 inches at the center of the grip. Negligible control position free play (less than 0.1 inch) was noted in the collective control system.

Control Positions in Trimmed Forward Flight

13. Control positions in trimmed, coordinated (ball-centered) forward flight were evaluated at the conditions listed in table 1. Test techniques are described in appendix D. Test results are presented in figures 4 and 5, appendix E. At a forward cg, the longitudinal control position moved forward with increasing airspeed throughout the tested airspeed range. At aft cg, the characteristics were similar at airspeeds greater than 55 knots calibrated airspeed (KCAS). Between 40 and 55 KCAS there was no position change with airspeed and between 30 and 40 KCAS, a control reversal existed where the longitudinal control position moved aft with increasing airspeed. The longitudinal, lateral, and directional control position changes with airspeed during level flight were minimal. Trim changes from level flight to autorotation or climbs were minor. The control positions in trimmed forward flight were similar to the OH-58A and are satisfactory.

Static Longitudinal Stability

14. Static longitudinal stability characteristics of the OH-58C were evaluated at the conditions listed in table 1. Test techniques are described in appendix D. The variations of control position with airspeed at constant collective are presented in figures 6 through 8, appendix E. In level flight at 52 KCAS, partial power descent at 76 KCAS, and autorotation at 75 KCAS, the aircraft exhibited positive static longitudinal stability. In level flight at 97 KCAS, static longitudinal stability was neutral for airspeeds below trim and positive but weak for airspeeds above trim. Moderate pilot compensation was required to establish and maintain desired airspeed (HQRS 4). This increased pilot workload to trim the aircraft will detract from mission tasks. The neutral static longitudinal stability of the OH-58C at cruise airspeed is a shortcoming. The static longitudinal stability characteristics of the OH-58C at 97 KCAS and an aft cg failed to meet the requirements of paragraph 3.2.10 of MIL-H-8501A, in that longitudinal control position stability with respect to airspeed was not positive.

Static Lateral-Directional Stability

15. Static lateral-directional stability characteristics of the OH-58C were evaluated at the conditions listed in table 1, using the techniques described in appendix D. Test results are presented in figure 9, appendix E.

16. Static directional stability, as indicated by the variation of directional control position with sideslip, was positive at all test conditions. Static directional stability was weaker at 51 KCAS than at 97 KCAS. The static directional stability of the OH-58C was essentially unchanged from OH-58A test results and is satisfactory.

17. Dihedral effect, as indicated by the variation of lateral control position with sideslip, was positive except at 97 KCAS with left sideslip angles greater than 10 degrees. At this point, there was an apparent weakening of the dihedral effect

in that with increasing sideslip angle beyond 10 degrees left, no additional lateral control input was required. This was not considered an objectionable characteristic because in normal operations the pilot will not fly at these large sideslip angles. The effective dihedral of the OH-58C was essentially unchanged from OH-58A test results and is satisfactory.

18. Side-force characteristics, as indicated by the variation of roll attitude with sideslip, were positive. At high airspeeds, the side force provided the pilot a good cue for out-of-trim condition. The side-force characteristics of the OH-58C were essentially unchanged from the OH-58A and are satisfactory.

Maneuvering Stability

19. Maneuvering stability was evaluated in left and right turns at the conditions listed in table 1. Test techniques are described in appendix D. Results of the maneuvering stability tests at a cg of FS 112.3 (aft) are presented in figure 10, appendix E. At 49 KCAS, the gradient of longitudinal control position with normal acceleration (g) was positive and essentially linear. At 96 KCAS the gradient was neutral to the test limit of 1.67g. Further testing was conducted at an aft cg of FS 111.2 (aft) (fig. 11, app E). At 86 KCAS the gradient of longitudinal control position with normal acceleration was positive. At 96 KCAS the gradient was positive up to a normal acceleration of approximately 1.2g, then became neutral and remained neutral to the test limit of 1.62g. The OH-58C maneuvering stability was degraded in comparison to the OH-58A. Although the neutral maneuvering stability in itself was not objectionable, it is a contributing factor to the undesirable pitch-up tendency discussed in paragraphs 23 and 27.

Dynamic Stability

20. The short-term longitudinal, lateral, and directional dynamic stability characteristics of the OH-58C were evaluated in level flight at conditions shown in table 1. Test techniques are described in appendix D. Short-term longitudinal and lateral responses were essentially deadbeat at 50 and 96 KCAS. Directional pulse inputs resulted in light to moderately damped Dutch-roll oscillations which subsided in 10 to 15 seconds or approximately 2 cycles. The short-term dynamic stability characteristics of the OH-58C were similar to the OH-58A and are satisfactory.

21. The long-term longitudinal dynamic stability characteristics of the OH-58C were evaluated in level flight at conditions shown in table 1. Test techniques are described in appendix D.

22. At a trim airspeed of 51 KCAS, the longitudinal long-term response was convergent but very lightly damped. Damping ratios (ζ) less than 0.05 were observed. The period of the oscillation was approximately 25 seconds. A typical time history of a longitudinal long-term response at 51 KCAS is presented in figure 12, appendix E. The oscillatory long-term response was readily excited in turbulent air and maneuvering flight at airspeeds between 40 to 60 knots indicated airspeed (KIAS) but was easily controlled (HQRS 3). The longitudinal long-term response at 51 KCAS was similar to the OH-58A and is satisfactory.

23. Following a release from 10 knots below a trim airspeed of 94 KCAS, airspeed returned through the original trim value and stabilized at approximately 100 KCAS. Following a release from 7 knots faster than the 94 KCAS trim airspeed, airspeed returned through trim and stabilized at approximately 85 KCAS (fig. 13, app E). Following a release from 13 knots faster than the 94 KCAS trim airspeed, the aircraft appeared to dig in and required considerable pilot compensation to arrest a divergent pitch-up. A time history (fig. 14) shows that pitch rate and cg normal acceleration linearly increased with time. This longitudinal pitch-up tendency at aft cg and cruise airspeed was uncomfortable to the pilot, was objectionable during maneuvering flight, and will be discussed further in paragraph 27.

24. Lateral-directional dynamic characteristics of the OH-58C were evaluated following releases from steady sideslips at the conditions shown in table 1. Test techniques are described in appendix D. The roll and yaw response was deadbeat following releases from right and left sideslip at 51 KCAS (fig. 15, app E). However, following all releases at 51 KCAS, the aircraft entered a shallow turn (4 to 8-degree bank angle). Following releases from right and left sideslips at 94 KCAS, moderately damped Dutch-roll oscillations were observed, which subsided in approximately 2 cycles (fig. 16). Roll to yaw ratio was approximately 1, and the oscillation period was approximately 5 seconds. A shallow turn (4 to 6-degree bank angle) was entered following releases from right sideslips at 94 KCAS. The lateral-directional dynamic stability characteristics of the OH-58C are satisfactory.

25. An evaluation of the spiral stability characteristics of the OH-58C was conducted in level flight at the conditions shown in table 1. Test techniques are described in appendix D. At 51 KCAS the spiral response was convergent, with a time to half amplitude of 13.5 seconds (fig. 17, app E). At 94 KCAS, the spiral response was also convergent with a time to half amplitude of 4.5 seconds (fig. 18). The spiral stability characteristics of the OH-58C are satisfactory.

26. The OH-58C was qualitatively evaluated during cyclic-only turns in both directions. No adverse yaw tendency was observed during these maneuvers, and the aircraft began turning in the proper direction following a momentary hesitation. The adverse yaw characteristics of the OH-58C are satisfactory.

Controllability

27. The longitudinal, lateral, and directional control response (maximum angular rate per inch of control input) and control sensitivity (maximum angular acceleration per inch of control input) of the OH-58C were evaluated in level flight at the conditions shown in table 1. Test techniques are described in appendix D. Test results are presented in figures 19 through 21, appendix E. The helicopter responded in the proper direction within the specification allowable time (0.2 second) for all control inputs. No objectionable mechanical or aerodynamic coupling was observed during any inputs. A summary of control response and sensitivity is presented in table 3. The values presented in the table are comparable to previous OH-58A test results except for longitudinal control sensitivity, which was approximately 10 deg/sec²/in. greater for the OH-58C. The angular rate responses for all control inputs except aft longitudinal were concave down from the time of control input to achieving maximum angular rate. Following aft

longitudinal control inputs, pitch rate and cg normal acceleration responses increased linearly until recovery became necessary. A time history of a typical aft longitudinal input is presented in figure 22, appendix E. The aircraft appeared to dig in and required immediate pilot control response to relieve the rapidly increasing normal acceleration. This pitch-up tendency was objectionable during maneuvering flight and required increased pilot compensation to control increased load factors following abrupt aft longitudinal cyclic control inputs (HQRS 5).

Table 3. Control Response and Sensitivity Summary.^{1,2}

Axis	Direction	Control Response (deg/sec/in.)	Control Sensitivity (deg/sec ² /in.)
Longitudinal	Forward	7.1	22.0
	Aft	9.0 ³	25.0
Lateral	Left	18.0	42.0
	Right	22.5	42.5
Directional	Left	15.0	44.5
	Right	18.0	44.5

¹Average gross weight: 3130 pounds; average cg location: FS 112.1 (aft); average density altitude: 7660 ft; average rotor speed: 354 rpm; trimmed airspeed: 94 KCAS; tests conducted in level flight;

²Based on maximum angular rate and acceleration achieved unless otherwise noted.

³Based on pitch rate achieved at 1 second following input.

The pitch-up tendency at aft cg and cruise airspeed is a shortcoming. The pitch rate and cg normal acceleration response following an aft longitudinal control step input at 94 KCAS and aft cg failed to meet the requirements of paragraph 3.2.11.1 of MIL-H-8501A, in that time histories of these responses did not remain concave down or achieve a maximum value.

Low-Speed Flight Characteristics

28. Low-speed forward, rearward, and sideward flight tests were conducted to determine control margins and flight characteristics of the OH-58C. The results indicate the ability of the OH-58C to hover in ground effect in varying wind conditions. Test techniques are described in appendix D.

29. The variation of pitch attitude and flight control positions with true airspeed for low-speed forward and rearward flight is presented in figure 23, appendix E. Stabilized forward flight was easily accomplished (HQRS 2), and the slight longitudinal control reversal between 15 and 35 knots was not objectionable. Steady, smooth rearward flight could not be satisfactorily performed because of excessive pitch and yaw oscillations. Pitch excursions of ± 2 to 3 degrees and yaw excursions of ± 2 to 8 degrees required considerable pilot compensation to control (HQRS 5). When hovering in tail wind conditions, the scout pilot will be required to devote most of his attention to aircraft control instead of mission requirements. The pitch and yaw oscillations in rearward flight are a shortcoming. The rearward flight characteristics of the OH-58C failed to meet the requirements of paragraph 3.2.1 of MIL-H-8501A, in that steady, smooth flight was not obtainable in rearward flight to 30 KTAS.

30. The variation of roll attitude and flight control positions with true airspeed for sideward flight is presented in figure 24, appendix E. Right sideward flight was easily stabilized from zero knots to the limit airspeed achieved (HQRS 2). The right sideward limit airspeed was 33 knots defined by the left directional control limit. At this airspeed, zero left directional control margin remained. Insufficient left directional control in right sideward flight is a shortcoming. During this test, the aircraft was tested at maximum gross weight and a low density altitude. The inadequate control margin situation will deteriorate at higher density altitudes. The right sideward flight characteristics of the OH-58C failed to meet the requirements of paragraph 3.3.2 of MIL-H-8501A, in that a sideward velocity of 35 knots was not achieved. The requirements of paragraph 3.3.6, as amended by reference 6, were also not met, in that there was zero control margin at the right sideward airspeed limit.

31. Smooth, steady left sideward flight between approximately 15 to 30 KTAS could not be satisfactorily performed because of excessive yaw oscillations. Yaw excursions of ± 3 to 9 degrees were observed which required extensive pilot compensation to control (HQRS 6). During hover in left crosswind conditions, the scout pilot will be required to devote most of his attention to aircraft control, and his tactical mission performance will be degraded. The yaw oscillations in left sideward flight between 15 and 30 KTAS are a shortcoming. This shortcoming has been reported in two previous evaluations of the OH-58A helicopter (refs 8 and 9, app A). The left sideward flight characteristics of the OH-58C failed to meet the requirements of paragraph 3.3.2 of MIL-H-8501A, in that smooth, steady flight was not achieved between approximately 15 and 25 KTAS.

Simulated Engine Failure

32. The response of the OH-58C following a simulated engine failure was evaluated at the conditions shown in table 1. Test techniques are described in appendix D. Time histories of aircraft response to simulated engine failure in level flight at 96 KCAS with maximum continuous power and in climbing flight at 53 KCAS with takeoff power are presented in figures 25 and 26, appendix E, respectively. The response of the aircraft following a simulated engine failure was generally characterized by rapid rotor speed decay, rapid left yaw, slight nose-down pitching (which increased with airspeed), and slight left roll.

33. Warning of the loss of engine power was provided first by the rapid left yaw, which was quickly sensed by the pilot and then by the low rotor speed aural warning system, which activated at approximately 332 rpm. At maximum level flight airspeed and maximum power climb test conditions, the rapid main rotor speed decay and left yaw of the aircraft were the critical response factors, which required collective and directional control inputs less than 2 seconds after throttle reduction. Following collective control reduction, main rotor speed was rapidly regained, and transition into stabilized autorotational descent was easily accomplished (HQRS 2). The aircraft response characteristics following a simulated engine failure were similar to the OH-58A and are satisfactory. The OH-58C failed to meet the requirements of paragraph 3.5.5.1 of MIL-H-8501A, in that yaw attitude change exceeded 10 degrees in less than 2 seconds following a sudden loss of power in level flight at 96 KCAS and in climbing flight with takeoff power at 53 KCAS.

HUMAN FACTORS

Cockpit Evaluation

34. A cockpit evaluation was conducted throughout the pilot training and flight test phases. The possibility of inadvertently activating the night vision goggle switch located on the pilot cyclic control grip was noted. Activation of this switch during daylight caused the warning and caution lights to dim to below daylight readability level. The possibility of inadvertent activation of the night vision goggle switch and resultant dimming of the warning and caution lights is a deficiency.

35. Six shortcomings were noted and should be corrected to improve the serviceability of the aircraft.

a. The transponder (AN/APX-100) was located on the lower console directly below and to the left of the pilot collective control. This position made it necessary for the pilot to reposition his head in order to see and set the code in the transponder. The location of the transponder display and controls is a shortcoming.

b. The force trim release button was located on the top right side of the cyclic control grip. This location required the pilot to reposition his hand on the cyclic control grip each time the force trim release button was depressed, causing extraneous inputs to the flight controls. The location of the force trim release button is a shortcoming.

c. Deflections in the turn needle were noted on the instrumented aircraft (SN 68-16850) each time the force trim release button was depressed. In some instances, the deflections were full scale. The electrical interference between the force trim release and the turn needle is a shortcoming.

d. The following shortcomings were noted on the noninstrumented aircraft (SN 69-16214):

- (1) The ammeter fluctuated ± 10 amperes during flight and on the ground.
- (2) The fuel gauge fluctuated ± 50 pounds during flight and on the ground.
- (3) The manual fuel control shutoff handle was not labeled for ON or OFF positions.

Night Evaluation

36. During the pilot training phase 1 hour was flown at night in clear, full moon conditions and the cockpit was evaluated for suitability and reflections. The instruments and gauges are well lit with no internal reflections. There were adequate provisions to dim all cockpit lights. Reflections from the lower console were noted in the left triangular windshield, but were not considered objectionable. No significant reflections from ground lights were noted in the cockpit.

SUBSYSTEM TESTS

37. A limited qualitative evaluation of an electromagnetic interference between the VHF-AM radio set (AN/ARC-115) and the CONUS navigation receiver (AN/ARC-123(V)) was conducted. The aircraft was flown on a simulated instrument navigation mission and a VOR approach was also performed. The aircraft was flown to track inbound to the Acton VOR (frequency 110.6, elevation 820 feet mean sea level (MSL)) at an altitude of 2000 MSL on the 060 degree radial. At approximately 25 nautical miles (NM) from the VOR, the VHF radio set was keyed to transmit on four VHF frequencies (131.0, 122.8, 127.3, and 121.9 Mhz). On every frequency tested, the bearing pointer in the heading bearing indicator (HBI) moved to the park position, and the course deviation indicator (CDI) indicated an unreliably weak signal or an equipment malfunction. When the transmitter was unkeyed, normal indications would return. The interaction occurred each time the VHF transmitter was keyed. When the aircraft was within 20 NM of the VOR, the transmitter had to be keyed for 12 seconds on 131.0 Mhz before any malfunction occurred. All other frequencies tested gave no indication of a malfunction. Within 15 NM of the VOR at 2000 feet MSL, no malfunctioning could be generated. Using the Acton VOR, a simulated VOR approach was flown and no indication of interference between the two radios occurred during the approach. The aircraft was then flown outbound on the 060 degree radial to 25 NM before a VHF transmission would cause a malfunction indication. The following NOTE should be included in the operator's manual.

NOTE

Operating the transmitter of the VHF-AM radio set (AN/ARC-115) may cause indications of an unreliably weak signal or an equipment malfunction on the course deviation indicator and the bearing pointer of the heading-bearing indicator.

CONCLUSIONS

GENERAL

38. The following conclusions were reached upon completion of testing.

- a. The most significant difference between the flying qualities of the OH-58C and the OH-58A was a pitch-up tendency at cruise airspeeds and aft cg.
- b. One deficiency and 12 shortcomings were identified.

DEFICIENCIES

39. The following deficiency was identified: possibility of inadvertent activation of the night vision goggle switch and resultant dimming of the warning and caution lights (para 34).

SHORTCOMINGS

40. The following shortcomings were identified and are listed in order of decreasing importance. Shortcomings annotated by an asterisk (*) are common to both OH-58A and OH-58C helicopters.

- a. Pitch-up tendency at cruise airspeed and aft cg (para 27).
- b. Insufficient left directional control in right sideward flight (para 30).
- *c. Yaw oscillations in left sideward flight between 15 and 30 KTAS (para 31).
- *d. Pitch and yaw oscillations in rearward flight (para 29).
- *e. Large longitudinal and lateral cyclic trim control displacement bands (para 10).
- f. Neutral static longitudinal stability at cruise airspeed (para 14).
- g. Location of the transponder display and controls (para 35).
- *h. Location of the force trim release button (para 35).
- i. Electrical interference between the force trim release and the turn needle (para 35).
- j. Fluctuating ammeter (para 35).
- k. Fluctuating fuel gauge (para 35).
- l. Lack of ON and OFF position labels on the manual fuel shutoff handle (para 35).

SPECIFICATION COMPLIANCE

41. The OH-58C failed to meet the following requirements of MIL-H-8501A:

a. Paragraph 3.2.1 - Steady, smooth flight was not obtainable in rearward flight to 30 KTAS (para 29).

b. Paragraph 3.2.10 - The longitudinal control position stability with respect to airspeed at a trim airspeed of 97 KCAS and aft cg was not positive (para 14).

c. Paragraph 3.2.11.1 - The time history of the pitch rate and cg normal acceleration response following an aft longitudinal step input at 94 KCAS and aft cg did not remain concave down or achieve a maximum value (para 27).

d. Paragraph 3.3.2 - A right sideward velocity of 35 KTAS was not achieved (para 30).

e. Paragraph 3.3.2 - Smooth, steady left sideward flight was not achieved between 15 and 35 KTAS (para 31).

f. Paragraph 3.5.5.1 - Yaw attitude change exceeded 10 degrees in less than 2 seconds following a sudden loss of power in level flight at 96 KCAS and in climbing flight with takeoff power at 53 KCAS (para 33).

42. The OH-58C failed to meet the following requirement of MIL-H-8501A as amended by the BHT detail specification: Paragraph 3.3.6 - Zero left directional control margin remained at the limit right sideward airspeed (para 30).

RECOMMENDATIONS

- 43. The deficiency reported in paragraph 39 should be corrected prior to operational deployment of the OH-58C.
- 44. The shortcomings reported in paragraph 40 should be corrected as soon as possible.
- 45. The following NOTE should be added to the operator's manual (para 37).

NOTE

Operating the transmitter of the VHF-AM radio set (AN/ARC-115) may cause indications of an unreliably weak signal or an equipment malfunction on the course deviation indicator and the bearing pointer of the heading-bearing indicator.

APPENDIX A. REFERENCES

1. Letter, AVSCOM, DRSAB-EQI, 26 September 1976, subject: Army Preliminary Evaluation of an OH-58C.
2. Message, AVRADCOM, DRDAV-EQI, 311845Z, October 1977, subject: Modification of Test Schedule of USAAEFA Project No. 76-11, PAE, OH-58C Helicopter Test Plan.
3. Test Plan, USAAEFA, Project No. 76-11-1, *Preliminary Airworthiness Evaluation, OH-58C Helicopter*, December 1977 (unpublished).
4. Technical Manual, TM 55-1520-235-10, *Operator's Manual, Army Model OH-58C Helicopter*, Final Draft, 29 April 1977.
5. Letter, AVRADCOM, DRDAV-EQI, 13 February 1978, subject: Airworthiness Release for USAVRADCOM/USAAEFA Project No. 76-11.
6. Detail Specification, Bell Helicopter Textron, No. 206-947-203, "OH-58C Helicopter, Interim Scout," 19 September 1975, with revision 2, 1 March 1977.
7. Military Specification, MIL-H-8501A, *Helicopter Flying and Ground Handling Qualities; General Requirements For*, 7 September 1961, with amendment 1, 3 April 1962.
8. Final Report, US Army Aviation Systems Test Activity, Project No. 68-30, *Airworthiness and Flight Characteristics Test, Production OH-58A Helicopter Unarmed and Armed with XM27E1 Armament Subsystem*, October 1970.
9. Final Report, USAAEFA Project No. 74-48, *Evaluation of an OH-58A Helicopter with an Allison 250-C20B Engine*, April 1975.
10. Final Report, USAAEFA Project No. 75-11, *Army Preliminary Evaluation, JOH-58A Helicopter with Low Reflective Paint and Infrared Countermeasure Exhaust System*, December 1975.
11. Final Report, USAAEFA Project No. 75-20, *Army Preliminary Evaluation, YOIH-58A Helicopter with a Flat-Plate Canopy*, December 1975.
12. Flight Test Manual, Naval Air Test Center, FTM No. 101, *Stability and Control*, 10 June 1968.
13. Final Report, USAAEFA Project No. 74-22, *Engineering Evaluation, JOH-58A Helicopter with an Automatic Engine Relight System*, June 1975.

APPENDIX B. DESCRIPTION

GENERAL

1. The OH-58C helicopter is a derivative of the OH-58A scout helicopter. A portion of the existing OH-58A fleet is being modified to the OH-58C configuration by BHT. The modification consists of 19 primary changes to the aircraft in addition to numerous minor hardware changes. Table 1 presents a list of the 19 primary changes.

2. Overall aircraft dimensions and general configuration of the OH-58C are unchanged from the OH-58A. Maximum takeoff gross weight of the OH-58C is 3200 pounds compared to 3000 pounds for the OH-58A. The OH-58C main and tail rotors are identical to those on the OH-58A. A general description of the OH-58C aircraft including operating procedures and limitations is presented in the operator's manual. Specific changes will be discussed in the following paragraphs.

ENGINE

3. The T63-A-720 turboshaft engine built by Allison Division of Detroit Diesel Corporation is installed in the OH-58C. The physical characteristics of this engine are similar to the T63-A-700 turboshaft engine which powers the OH-58A. The T63-A-720 has an uninstalled sea-level, standard-day maximum rating of 420 shp and a continuous rating of 370 shp.

4. The engine is equipped with an automatic relight system which provides automatic engine reignition in the event of engine flameout. The system consists of a control box, bleed air pressure sensing line, and electrical connectors. In the event of bleed air pressure loss resulting from engine flameout, the system causes electrical power to be applied to the ignitor plug to resume fuel combustion. Activation of the system is indicated by the illumination of a light on the cockpit instrument panel. An additional system description is presented in reference 13, appendix A.

MAIN TRANSMISSION

5. Numerous bearing design and metallurgical improvements are incorporated in the main transmission. Although the power output limit of the transmission is unchanged from the OH-58A, the transmission's dry run capability and survivability in the event of loss of lubrication fluid is improved.

6. Improved main transmission pylon support fittings are installed to enhance the crashworthiness of the main transmission support structure. These fittings, which are used to fasten the main transmission pylon support link to the roof beam of the helicopter, are fabricated from a stronger steel alloy.

7. A main transmission oil pressure gauge on the cockpit instrument panel allows direct reading of transmission oil pressure.

Table 1. OH-58C Primary Changes.

<p>T63-A-720 engine</p> <p>Flat glass cockpit canopy</p> <p>Tail rotor drive shaft cover</p> <p>Improved tail rotor drive shaft bearings</p> <p>Infrared suppression engine exhaust</p> <p>Infrared suppression engine cowl</p> <p>Low reflective fuselage paint</p> <p>Automatic engine relight system</p> <p>Vulnerability reduced flight controls</p> <p>Controllable landing light</p>	<p>Improved main transmission bearings</p> <p>Improved main transmission pylon support fittings</p> <p>Transmission oil pressure gauge</p> <p>Reconfigured instrument panel</p> <p>Improved cabin air distribution system</p> <p>AN/APR-39 radar warning</p> <p>AN/APR-123 CONUS navigation</p> <p>AN/APR-209 radar altimeter</p> <p>YG-1054 proximity warning</p>
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TAIL ROTOR DRIVE SHAFT

8. A cover is installed over the tail rotor drive shaft to aid in keeping dust and dirt from the tail rotor drive shaft bearings. The cover, illustrated in photo 1, extends from the engine oil reservoir cowling to just forward of the tail rotor gearbox.

9. New tail rotor drive shaft bearings and improved rubber seals are installed on the OH-58C. The new bearing, which must be periodically lubricated, and the improved rubber seal increase the reliability of the tail rotor drive shaft.

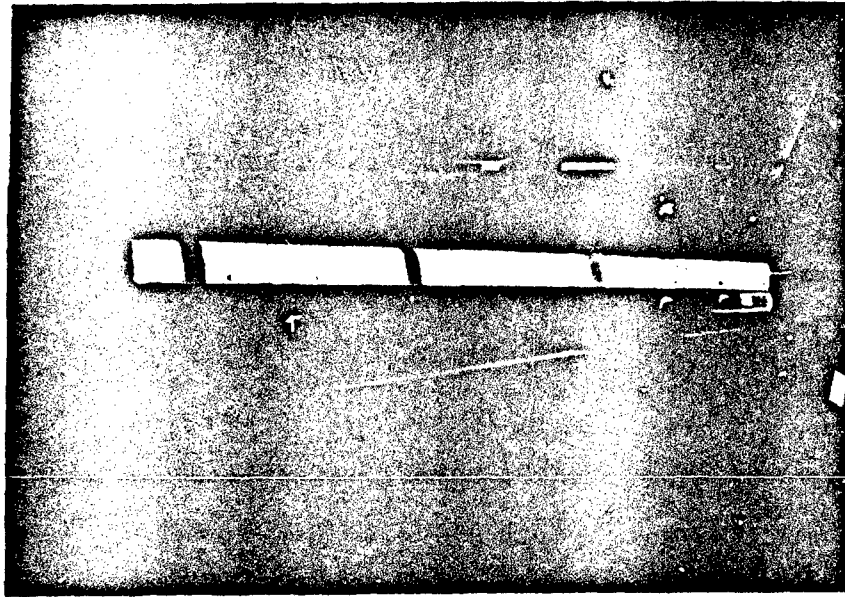


Photo 1. Tail Rotor Drive Shaft Cover.

FUSELAGE

10. A four-panel flat plate canopy is installed on the OH-58C in place of the bubble canopy on the OH-58A. Two rectangular forward panels and two triangular side panels are fabricated from stretched acrylic and riveted in place. Photos 2 and 3 illustrate the flat plate canopy installation. Additional descriptions are presented in reference 11, appendix A.

11. Infrared suppressive engine exhaust ducts are installed on the OH-58C. These exhaust ducts, illustrated in photo 4, incorporate cooling fins and serve to cool the engine exhaust gases before venting to the atmosphere.

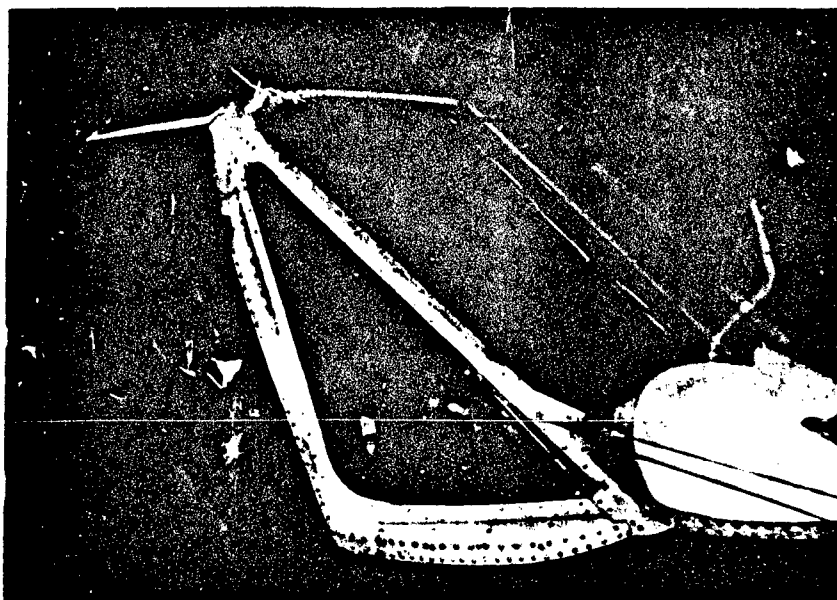


Photo 2. Flat Plate Canopy-Right Side.

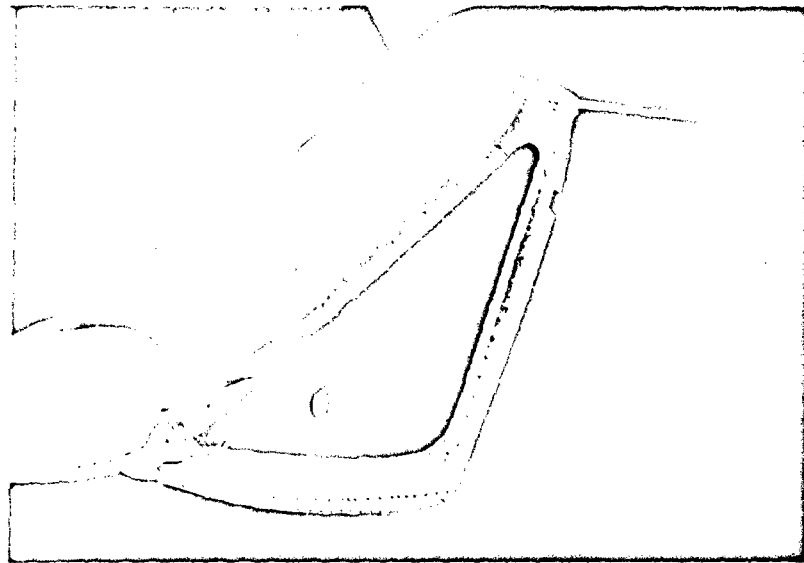


Photo 3. Flat Plate Canopy-Left Side.

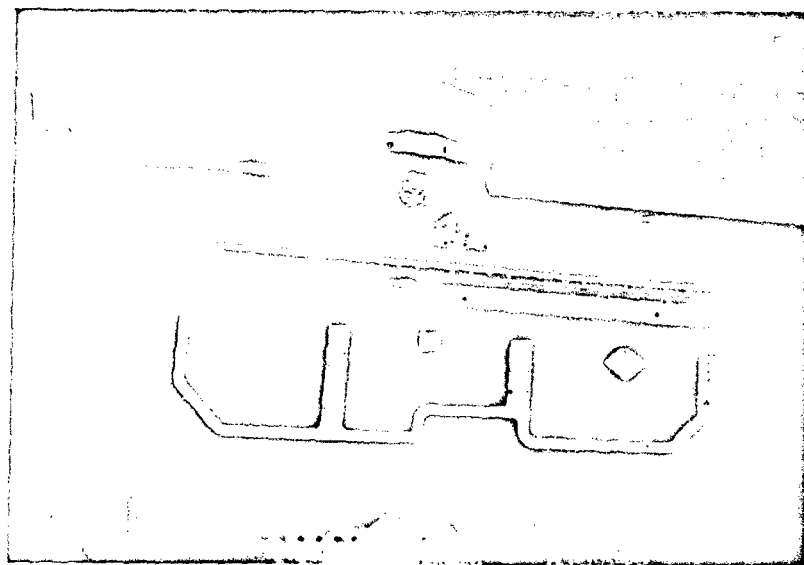


Photo 4. Exhaust Ducts and Screen Port.

12. The engine cowl of the OH-58C is redesigned to suppress infrared radiation from the engine area and provide improved engine cooling. Two screened ports are located on the top panel of the engine cowl forward of the engine exhaust ducts. The side panels of the engine cowl are modified as illustrated in photo 4 to incorporate a heat elimination tunnel.

13. The external fuselage of the OH-58C is painted with low reflective paint. This paint, which conforms to MIL-L-46159, reduces glare from the aircraft structure.

14. A controllable landing light is installed on the OH-58C. The light is controllable in elevation and azimuth and is similar to a UH-1 searchlight. A four-way momentary thumb switch is installed on the collective control head to provide control of the landing light position (fig. 1).

FLIGHT CONTROLS

15. The cyclic and collective controls of the OH-58C are essentially unchanged from the OH-58A. Four structural changes have been incorporated to reduce the vulnerability of the cyclic and collective control systems to ballistic damage. The longitudinal cyclic, lateral cyclic, and collective vertical push-pull tubes located in the enclosed column behind the front cockpit seats have been enlarged in diameter. The cyclic yoke beneath the front cockpit seats has also been enlarged.

16. A backup tail rotor control system is installed to provide redundant tail rotor control. A diagram of the primary and backup tail rotor control systems is presented in figure 2. The backup system consists of a push-pull cable located in a rigid housing. The push-pull cable is connected into the primary tail rotor control system at a forward bell crank between the pilot and the copilot pedals and at an aft bell crank below the tail rotor gearbox. The backup tail rotor control system functions with the primary system during normal operation. In the event of a control jam, the primary system is disconnected at each end by an electromechanical disconnect link, enabling the tail rotor to be controlled solely by the backup control system. The two disconnect links in the primary system are activated by a toggle switch located on the pilot collective control stick (fig. 3). Lights on the instrument panel indicate if the primary tail rotor system is jammed and if the disconnect switch has been activated.

INSTRUMENT PANEL

17. The OH-58C instrument panel is enlarged to incorporate additional instruments, indicators, and avionics controls. A diagram of the instrument panel is presented in figure 4. Four principal new items of avionics equipment have been added and are listed below. A more detailed description including operational procedures for these items is presented in the operator's manual.

- a. AN/APR-39 radar warning set.
- b. AN/APN-123 CONUS navigation receiver.
- c. AN/APR-209 radar altimeter.
- d. YG-1054 proximity warning device.

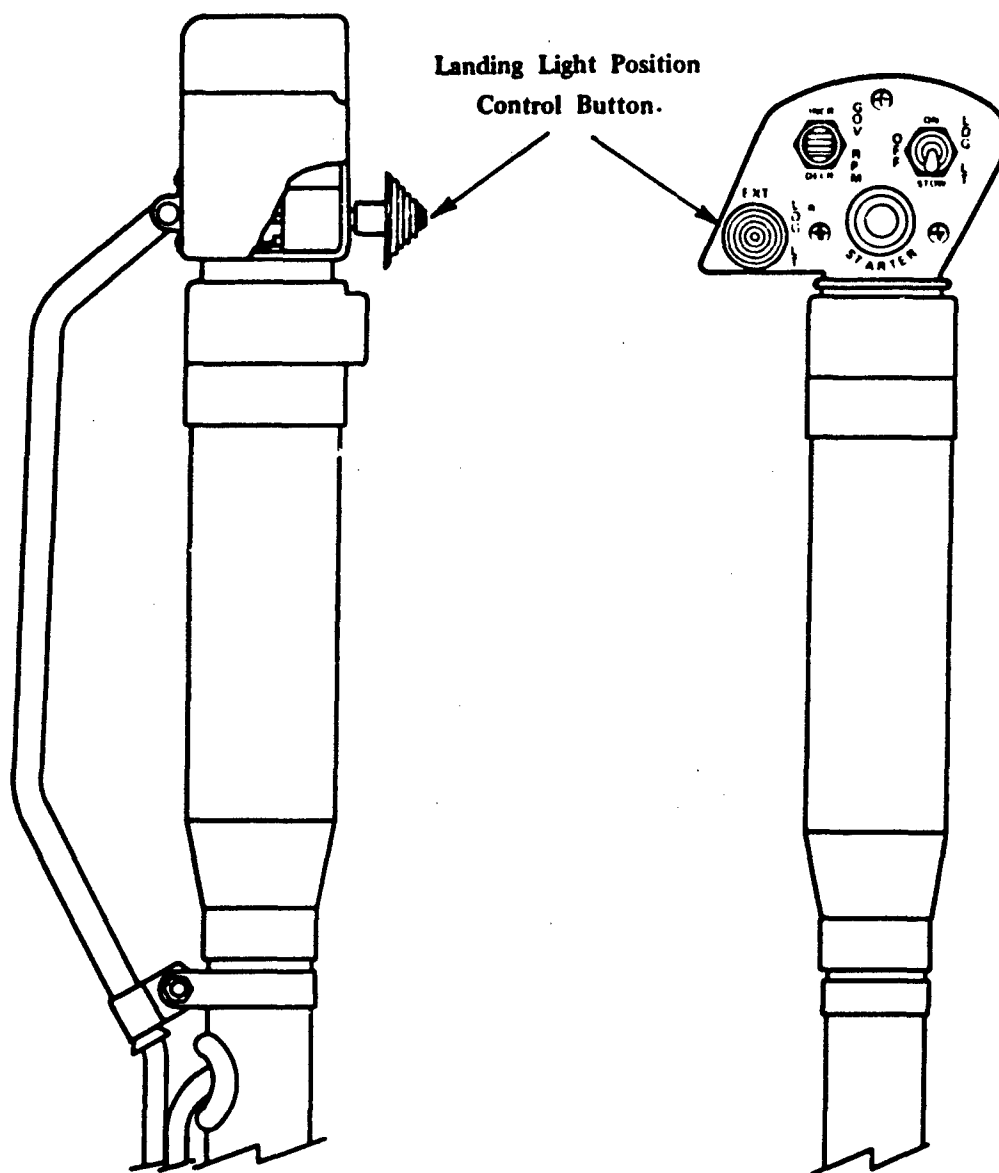


Figure 1. Collective Control Head.

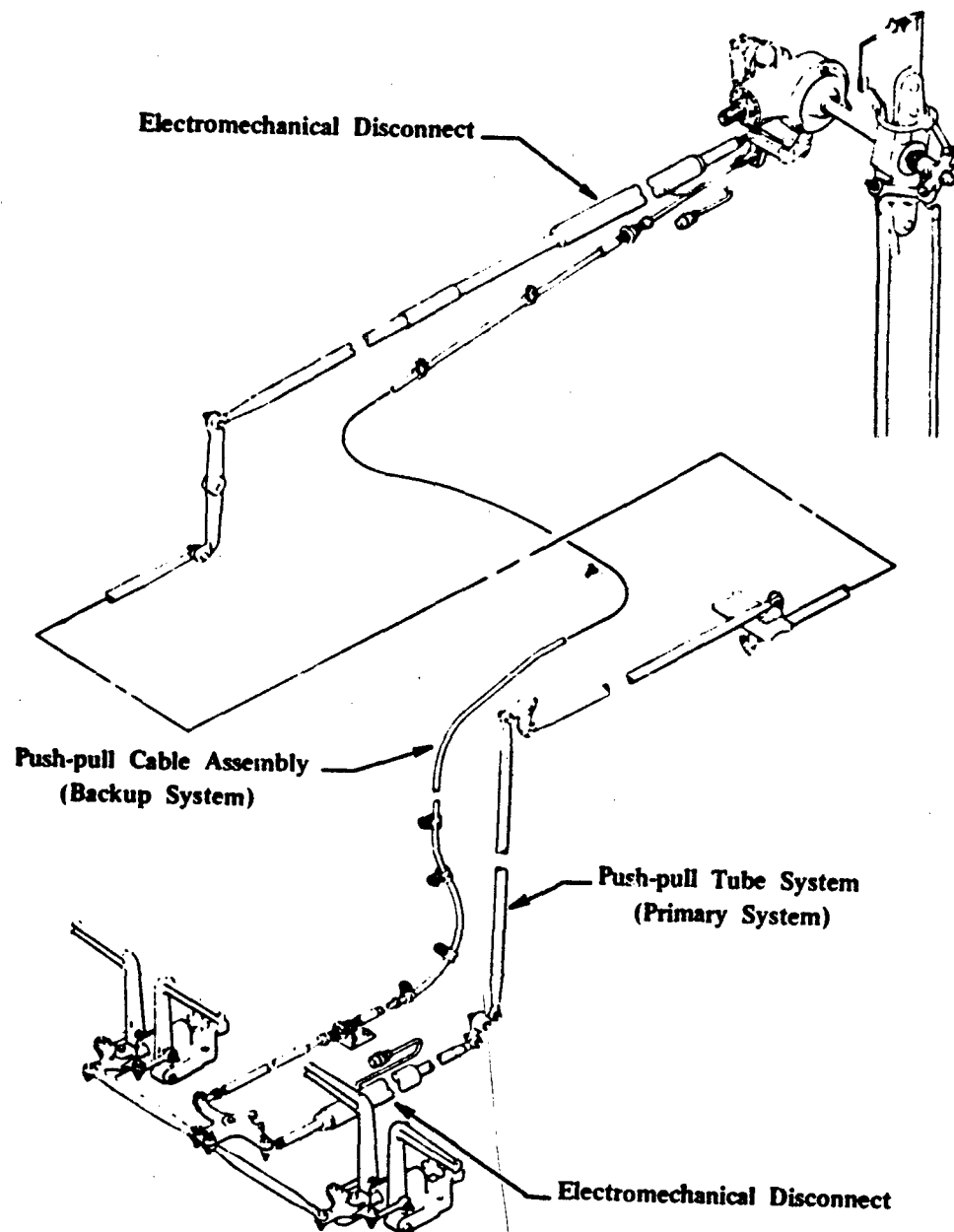


Figure 2. Tail Rotor Control System.

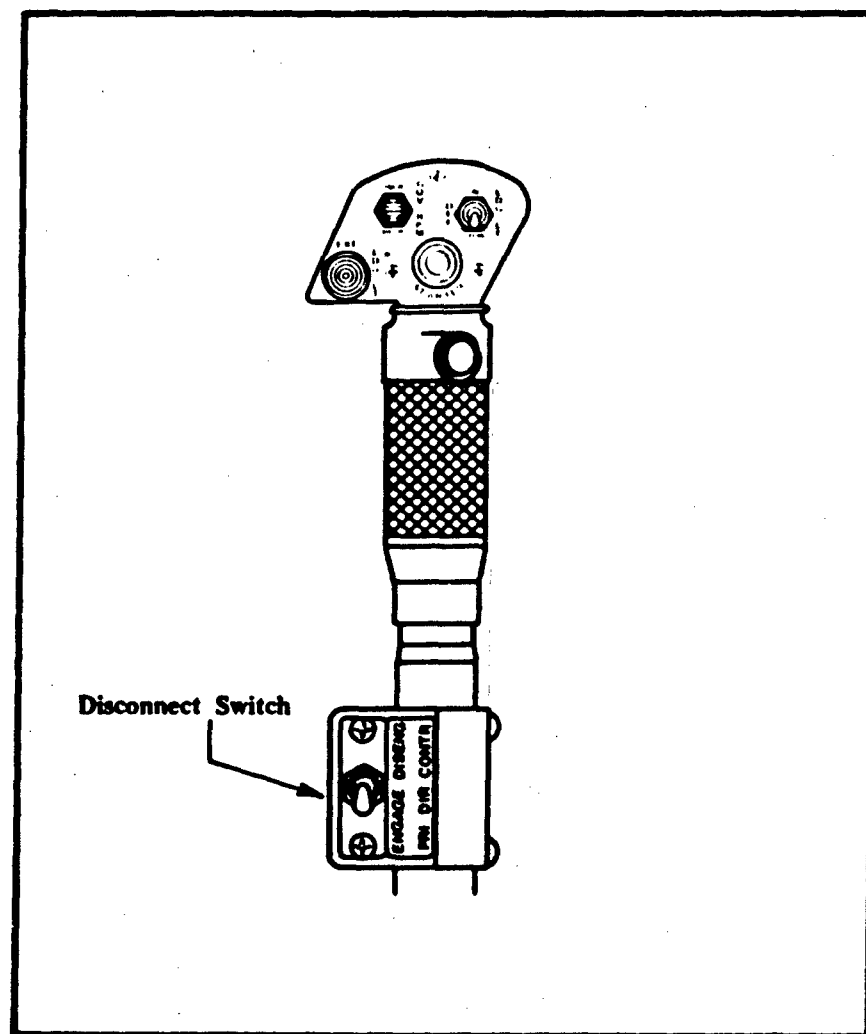


Figure 3. Tail Rotor Primary Control Disconnect Switch.

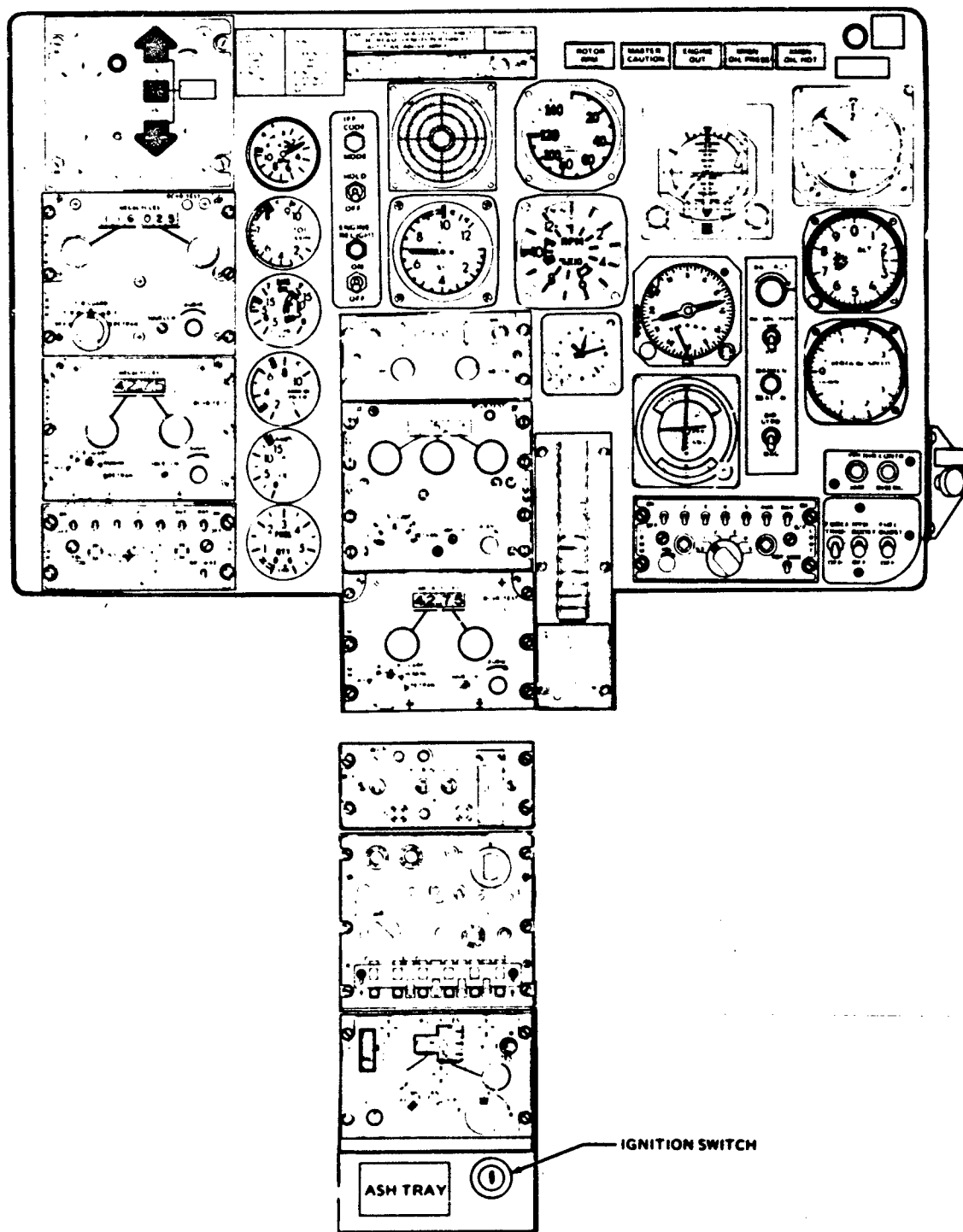


Figure 4. OH-58C Instrument Panel.

18. All instruments, indicators, and control panels on the instrument panel are internally lighted with red light. In addition, the engine and flight instruments employ low-light level symbology.

HEATING/DEFROSTING SYSTEM

19. Two ventilating and defogging blowers are installed in the nose of the OH-58C. These blowers improve the flow of bleed air heat to the windshields for defrosting and improve the flow of ventilating air in the cockpit.

CONTROL SYSTEM RIGGING CHECK

20. A rigging check was performed in the aircraft prior to testing. Rigging measurements were taken in accordance with applicable maintenance procedures and were within acceptable limits.

WEIGHT AND BALANCE

21. Prior to testing, the aircraft gross weight and longitudinal and lateral cg were determined using calibrated scales. The aircraft was weighed empty in the clean configuration with instrumentation installed.

22. The fuel loading for each test flight was determined prior to engine start and following engine shutdown. A calibrated external sight gauge was used to determine fuel volume and specific gravity was measured using a hydrometer. Fuel used in flight was recorded by a test fuel-used system and verified with the pre- and postflight sight gauge reading.

23. Aircraft gross weight and cg were controlled by ballast installed at various locations in the aircraft.

FLIGHT LIMITATIONS

24. During this evaluation, the OH-58C was flown within the limitations prescribed in the operator's manual as amended by the airworthiness release. The amended limitations are presented in the following paragraphs.

a. The cg limitations were as depicted in figure 5.

b. The sideslip angle limitations were as depicted in figure 6.

c. The standard load factor limitations were as depicted in figure 7. The data in figure 7 are based on sea-level, standard-day conditions. Load factors corrected for test density altitude and gross weight conditions were computed using the following equation:

$$N_z = N_z' \sigma \frac{3200}{GW}$$

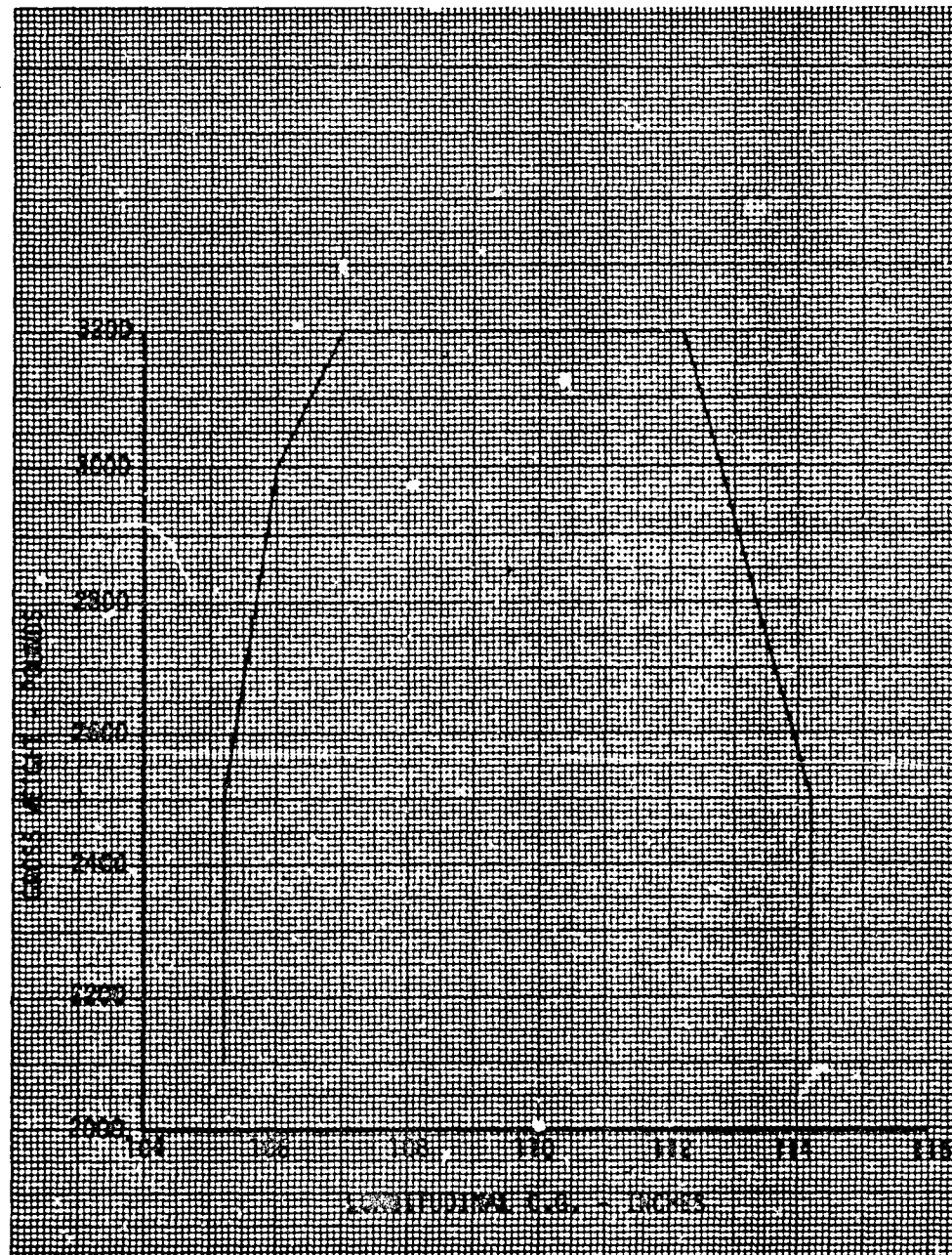


Figure 5. Airworthiness Release Center of Gravity Limitations.

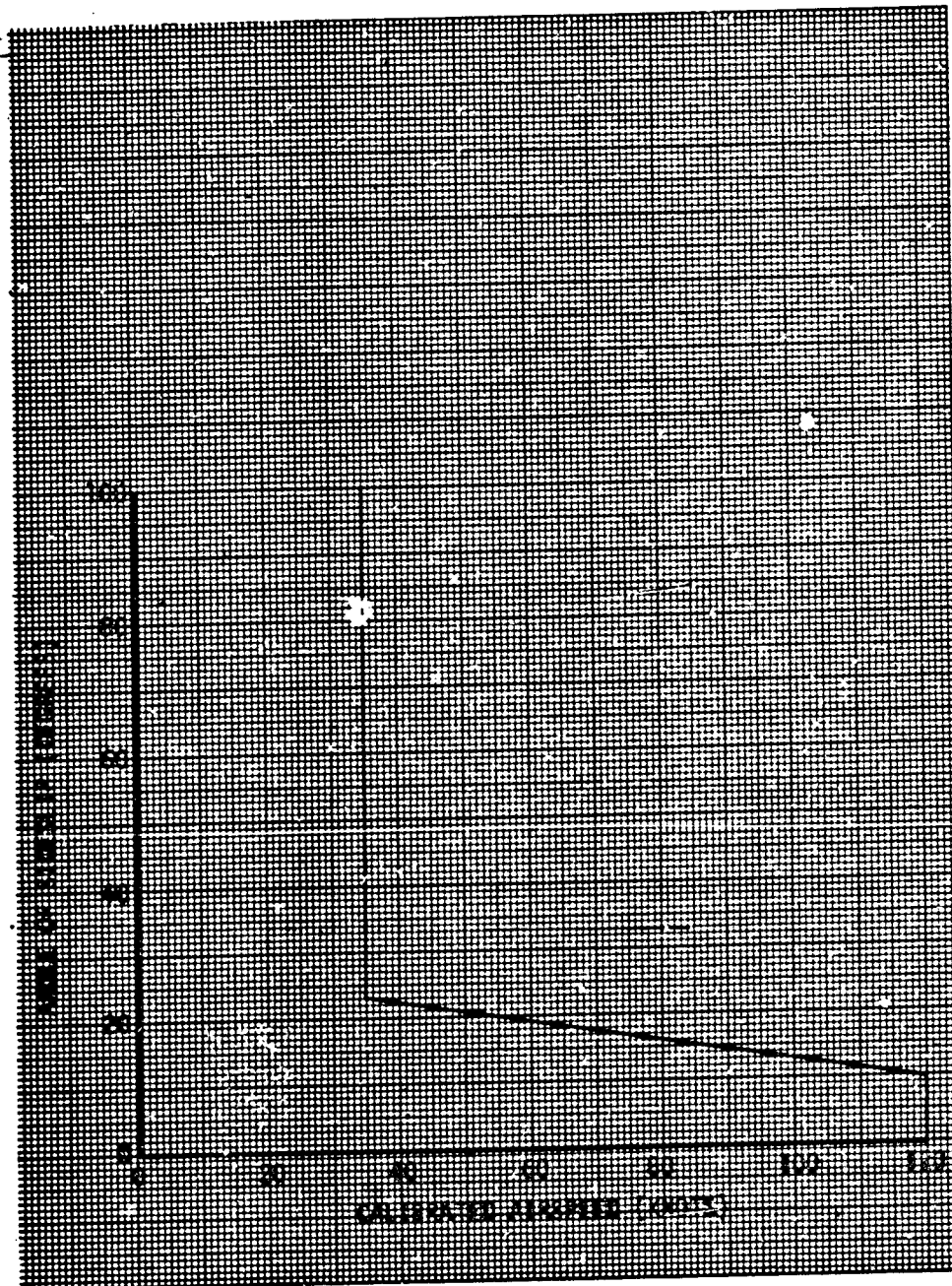


Figure 6. Airworthiness Release Sideslip Angle Limitations.

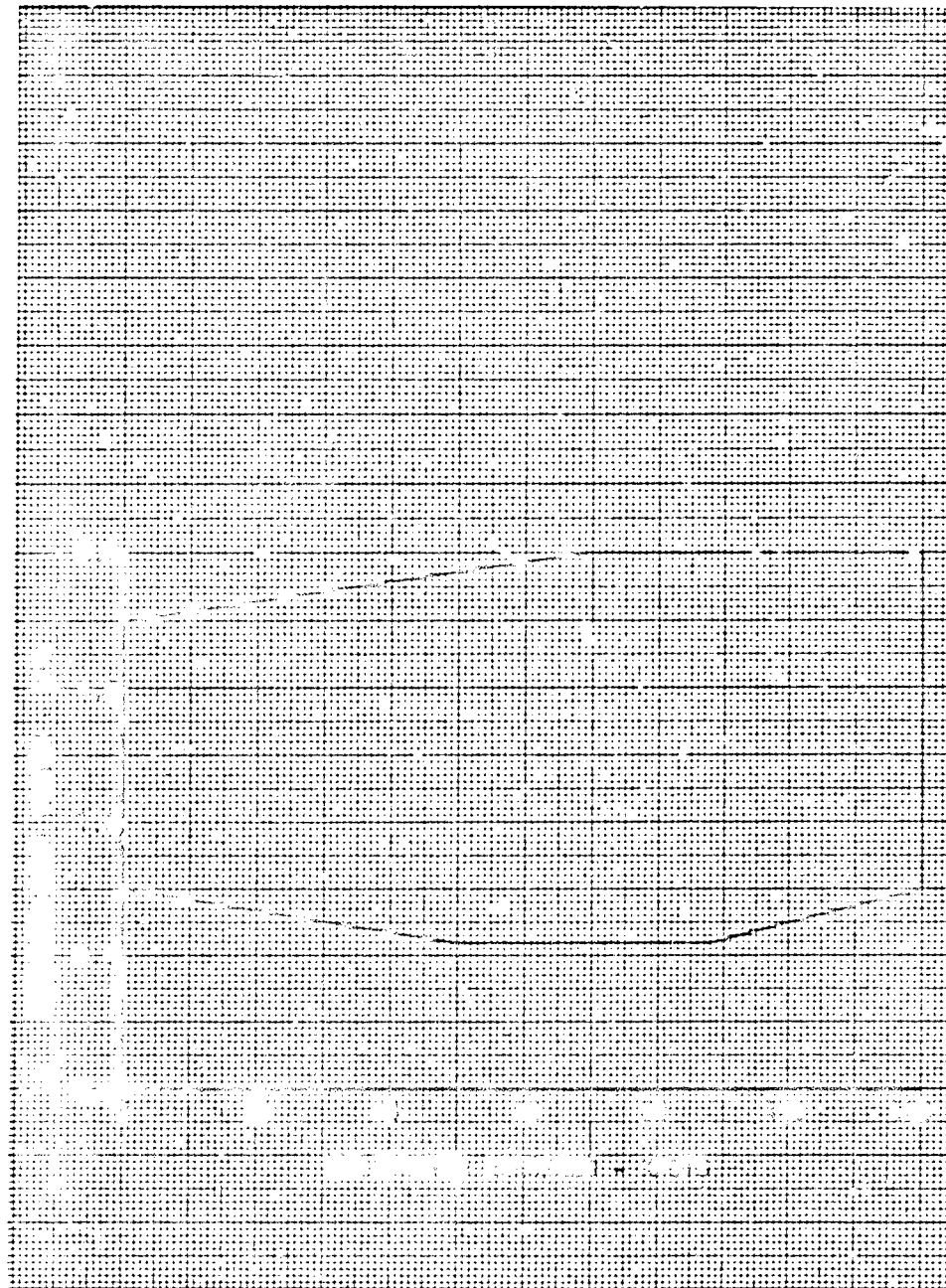


Figure 7. Airworthiness Release Standard Load Factor Limitations.

Where:

N_z = Corrected load factor

N_z' = Standard load factor

σ = Test altitude density ratio

GW = Aircraft gross weight

d. The airspeed calibration curve for the test aircraft boom system was as depicted in figure 8.

e. The following airframe and engine limitations applied to the test aircraft:

- (1) Maximum takeoff gross weight 3250 pounds
- (2) Main rotor speed
 - Power on continuous 98 to 100% (347 to 354 rpm)
 - Power on transient 93 to 102% (330 to 361 rpm)
 - Power off continuous 93 to 110% (330 to 390 rpm)
 - Power off transient 86 to 116% (304 to 411 rpm)
- (3) Engine output shaft speed
 - Maximum continuous 100% (6180 rpm)
 - Maximum transient (15 sec) 102%
- (4) Engine torque
 - Maximum continuous 85%
 - Maximum transient (5 min) 100% (269.4 ft-lb)
- (5) Engine gas generator speed
 - Maximum continuous 105% (100% = 50,970 rpm)
 - Maximum transient (15 sec) 106%
- (6) Engine turbine gas temperature
 - Maximum continuous 738°C
 - 30 minute limit 738 to 810°C
 - Starting transient (10 sec) 810 to 927°C
 - Acceleration transient (6 sec) 810 to 843°C
- (7) Main rotor blade flap angle (maximum) 85%

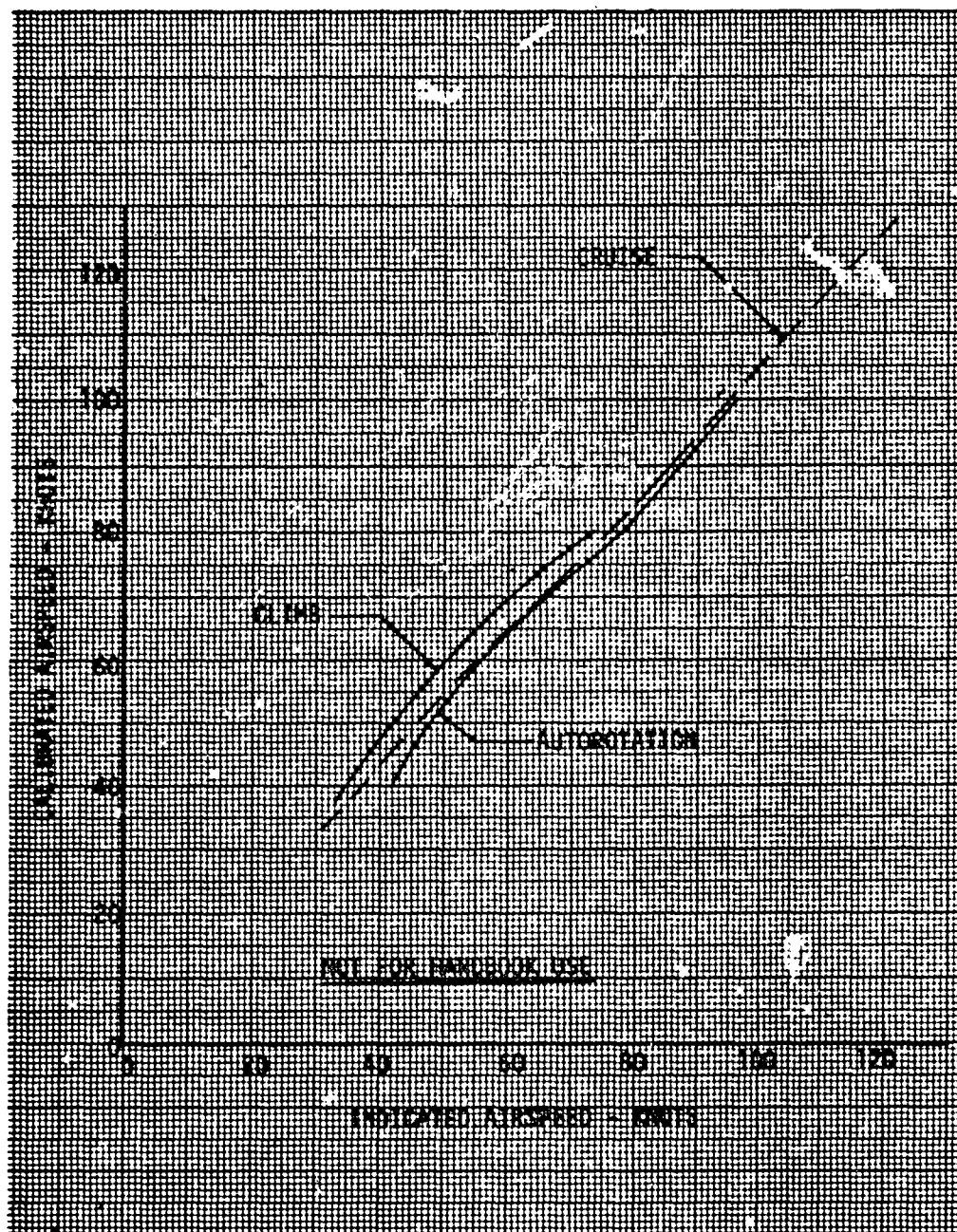


Figure 8. Airworthiness Release Airspeed Calibration.

f. The main rotor flapping angle indicator was required to be operational for all flights at forward cg and high gross weight.

g. The instrument light circuit breaker on the overhead console was saftied in the out position to deactivate the night vision goggle switch located on the pilot cyclic control head.

h. The following emergency procedure applied to the test aircraft:

If the XMSN OIL HOT warning light illuminates, accomplish a power-on approach and land immediately. If the XMSN OIL PRESS warning light illuminates, immediately check the transmission oil pressure gage and, based on the indicated pressure, take one of the following actions:

- (1) Exceeds 20 psi: Reduce power and land at nearest maintenance facility.
- (2) Less than 20 psi: Land with power immediately.

APPENDIX C. INSTRUMENTATION

1. The test instrumentation was installed, calibrated, and maintained by BHT. Data were obtained from calibrated instrumentation and were recorded on magnetic tape and/or displayed in the cockpit. The data acquisition system consisted of various transducers, signal conditioning units, frequency multiplexing (FM) technique, and a 1-inch, 14-track Inter-Range Instrumentation Group (IRIG) intermediate band recorder. Various specialized indicators displayed data to the pilot and engineer on board the aircraft continuously during the flight. A flight test boom was mounted on the nose of the aircraft with the following sensors: swiveling pitot-static head, sideslip vane, angle-of-attack vane, and total temperature sensor.

2. Specialized and/or calibrated cockpit monitored parameters are listed below.

- Airspeed (boom)
- Airspeed (ship's system)
- Altitude (boom)
- Altitude (radar)
- Angle of sideslip
- Cg normal acceleration
- Control position:
 - Longitudinal
 - Lateral
 - Directional
 - Collective
- Engine torque pressure (digital)
- Free air temperature
- Fuel used (totalizer)
- Gas generator speed (ship's system)
- Main rotor flapping angle
- Rate of climb/descent (boom)
- Rotor speed
- Turbine outlet temperature (ship's system)
- Event switch
- Instrumentation controls
- Record counter

3. Parameters recorded on tape were as follows:

- Airspeed (boom)
- Altitude (boom)
- Angle of attack
- Angle of sideslip
- Cg normal acceleration
- Control position:
 - Longitudinal
 - Lateral
 - Directional
 - Collective
 - Throttle

Aircraft attitude and angular velocity:

Pitch

Roll

Yaw

Engine torque pressure

Gas generator speed

Main rotor flapping angle

Rotor speed

Outside air temperature

Pilot/engineer event

Time code

Record number

APPENDIX D. TEST TECHNIQUES AND DATA ANALYSIS METHODS

TEST TECHNIQUES

1. Conventional test techniques were used during handling qualities tests. Detailed descriptions of test techniques are contained in reference 12, appendix A. Ball-centered trim was used as a reference for all flight tests. Handling qualities ratings were assigned in accordance with the handling qualities rating scale presented in figure 1.

Control System Characteristics

2. These tests were conducted on the ground with hydraulic and electrical power provided by ground power units. A hand-held force gauge was used to measure the force required to move the cyclic control incremental displacements to the limits of travel in four directions. Hysteresis was checked by taking measurements in the increasing and decreasing force direction.

Control Positions in Trimmed Forward Flight

3. These tests were conducted by first trimming the aircraft in coordinated flight at the selected airspeed and flight condition, and then recording the control positions. Altitude was allowed to vary (± 500 feet of the trim altitude) in climbs, dives, and autorotation.

Static Longitudinal Stability

4. These tests were accomplished by first trimming the aircraft in coordinated flight at the desired trim airspeed, then with collective control fixed, the helicopter was displaced from trim and stabilized at incremental airspeeds greater and less than the trim airspeed. Data were recorded at each stabilized airspeed.

Static Lateral-Directional Stability

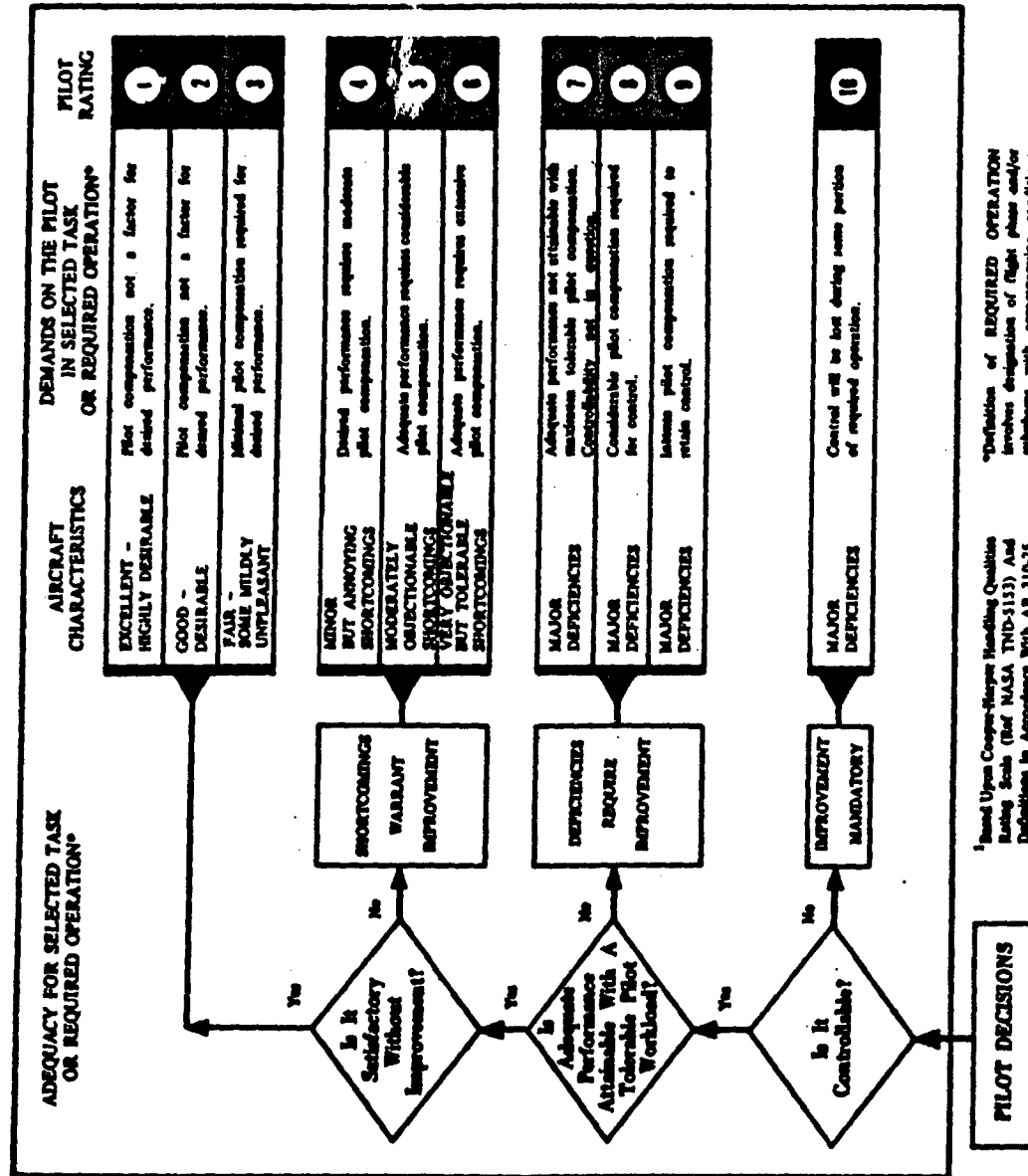
5. These tests were conducted by trimming the aircraft in coordinated flight at the desired conditions. With collective control fixed, the aircraft was then stabilized at incremental sideslip angles up to limit sideslip on both sides of trim while maintaining a steady heading at the trimmed airspeed.

Maneuvering Stability

6. These tests were conducted by first trimming the aircraft in coordinated flight at the desired condition, and then incrementally increasing load factor during steady turns (in both directions). Airspeed and collective control position were held constant while altitude was allowed to vary (± 500 feet of trim altitude).

Dynamic Stability

7. These tests consisted of evaluating both the short-term and long-term responses of the aircraft. Lateral-directional dynamic and spiral stability characteristics were



*Based Upon Cooper-McCoy Handling Qualities Rating Scale (Ref NASA TN-D-3133) And Definitions In Accordance With AR 310-25.

*Definition of REQUIRED OPERATION involves designation of flight phase and/or altitudes with accompanying conditions.

Figure 1. Handling Qualities Rating Scale.

also investigated. Short-term testing was accomplished by forward and aft longitudinal, right and left lateral, and right and left directional control pulse inputs. The pulse input was obtained by rapidly displacing the control approximately 1 inch, holding for 0.5 second, then rapidly returning to the trim position and holding until corrective action became necessary or aircraft motions were damped. All controls other than the input control remained fixed during the test. Long-term longitudinal characteristics were evaluated by displacing the aircraft from the trim airspeed approximately 10 knots. Both slow starts and fast starts were utilized. The slow-start technique consisted of reducing airspeed below the trim value using cyclic control, then returning the cyclic control to its original trim position using a control fixture and observing the resulting aircraft response. The fast-start technique was similar except that airspeed was increased above the original trim value. Lateral-directional dynamic stability tests were conducted by observing the aircraft response following releases from steady sideslips. The aircraft was first trimmed in ball-centered, wings-level flight and then a steady-heading sideslip was established. The flight controls were then rapidly returned to the original level flight trim positions, and the resulting aircraft response observed. Spiral stability tests were conducted by first trimming the aircraft in level flight at the desired condition, and then a 15-degree left or right bank angle was introduced using directional controls only. The directional controls were then returned to their original position for trimmed level flight, and aircraft response was observed.

Controllability

8. These tests were conducted by first trimming the aircraft in level flight at the desired condition, and then step control inputs of different magnitude were applied about the pitch, roll, and yaw axes. Each control was rapidly displaced, and then held firmly against a rigid fixture until the maximum rate was reached or until recovery became necessary.

Low-Speed Flight Characteristics

9. Testing was accomplished by flying the aircraft at a constant skid height of 10 feet in sideward (left and right), rearward, and forward flight in steady wind less than 6 knots. Wind azimuths of zero, 90, 180, and 270 degrees relative to the aircraft nose were used. Tests were flown at 5-knot increments up to a maximum airspeed of 40 knots forward and to the specification sideward and rearward airspeed limits. A ground pace vehicle with a calibrated speed system (5th wheel) was used as an airspeed reference during these tests.

Simulated Engine Failure

10. These tests were conducted by first stabilizing the aircraft at the desired trim flight condition and then simulating engine failure by rapidly reducing the throttle to engine idle. Controls were held fixed for 2 seconds after the power reduction or until recovery was necessary. The aircraft was then stabilized in autorotational descent.

DATA ANALYSIS METHODS

Indicated Airspeed and Pressure Altitude

11. Airspeed, static pressure, and total temperature were measured from sensors mounted on a flight test boom installed on the nose of the aircraft. The output signals were recorded on magnetic tape, and the following expressions were used to calculate the parameters:

- a. Indicated airspeed corrected for instrument error (V_{ic}):

$$V_{ic} = a_o \left\{ 5 \left[\left(\frac{q_{c_{ic}}}{P_{a_o}} + 1 \right)^{2/7} - 1 \right] \right\}^{1/2} \quad (1)$$

- b. Indicated pressure altitude corrected for instrument error (HP_{ic}):

$$HP_{ic} = \left[1 - \left(\frac{P_{a_{ic}}}{P_{a_o}} \right)^{1/5.255863} \right] / 6.8755856E-06 \quad (2)$$

Where:

V_{ic} = Indicated airspeed corrected for instrument error (kt)

a_o = Speed of sound at standard day, sea level = 661.483 kt

$q_{c_{ic}}$ = Indicated differential pressure corrected for instrument error (in. Hg)

P_{a_o} = Atmospheric pressure at standard day, sea level = 29.92126 in. Hg

HP_{ic} = Indicated pressure altitude corrected for instrument error (ft)

$P_{a_{ic}}$ = Indicated pressure altitude corrected for instrument error (in. Hg)

Calibrated Airspeed

12. BHT performed and furnished the airspeed position error correction for the boom pitot-static system. Calibrated airspeed was determined from the following equation:

$$V_{cal} = V_{ic} + \Delta V_{pc} \quad (3)$$

Where:

V_{cal} = Calibrated airspeed (kt)

ΔV_{pc} = Airspeed position error correction (kt)

Corrected Pressure Altitude and Altitude Position Error

13. HP_{ic} was corrected for altimeter position error by using ΔV_{pc} . The assumption was made that a pressure position error (ΔP_p) was produced entirely at the static source. Since both airspeed and altitude systems utilize the same static source, the following relationships were used:

$$\Delta P_p = 1.4 P_{a_o} \left(\frac{V_{ic}}{a_o} \right) \left[1 + 0.2 \left(\frac{V_{ic}}{a_o} \right)^2 \right]^{2.5} \frac{\Delta V_{pc}}{a_o} \quad (4)$$

$$+ 0.7 P_{a_o} \left[1 + 0.2 \left(\frac{V_{ic}}{a_o} \right)^2 \right]^{1.5} \left[1 + 1.2 \left(\frac{V_{ic}}{a_o} \right)^2 \right] \left[\frac{\Delta V_{pc}}{a_o} \right]^2$$

$$P_a = P_{a_{ic}} - \Delta P_p \quad (5)$$

$$H_p = \left[1.0 - \left(\frac{P_a}{P_{a_o}} \right)^{1/5.255863} \right] / 6.8755856E-06 \quad (6)$$

Where:

ΔP_p = Pressure position error (in. Hg)

P_a = Atmospheric pressure at corrected altitude (in. Hg)

HP = Corrected pressure altitude (ft)

Static Temperature

14. Static temperature was obtained by correcting the measured total temperature for temperature rise due to compressibility. The following assumptions were made.

- The temperature recovery factor is equal to 1.
- The equivalent airspeed is equal to calibrated airspeed.

The following expressions were used:

$$T_{Tic} = OAT_{ic} + 273.15 \quad (7)$$

$$T_a = \frac{T_{Tic}}{1 + \frac{\rho_o [V_{cal}] (1.6878)}{P_a (70.73)}} \quad (8)$$

$$OAT = T_a - 273.15 \quad (6)$$

Where:

OAT_{ic} = Indicated ambient temperature corrected for instrument error ($^{\circ}C$)

T_{Tic} = Indicated temperature corrected for instrument error (K°)

T_a = Static temperature (K°)

ρ_o = Air density at standard-day sea-level (slugs/ft³)

OAT = Static temperature ($^{\circ}C$)

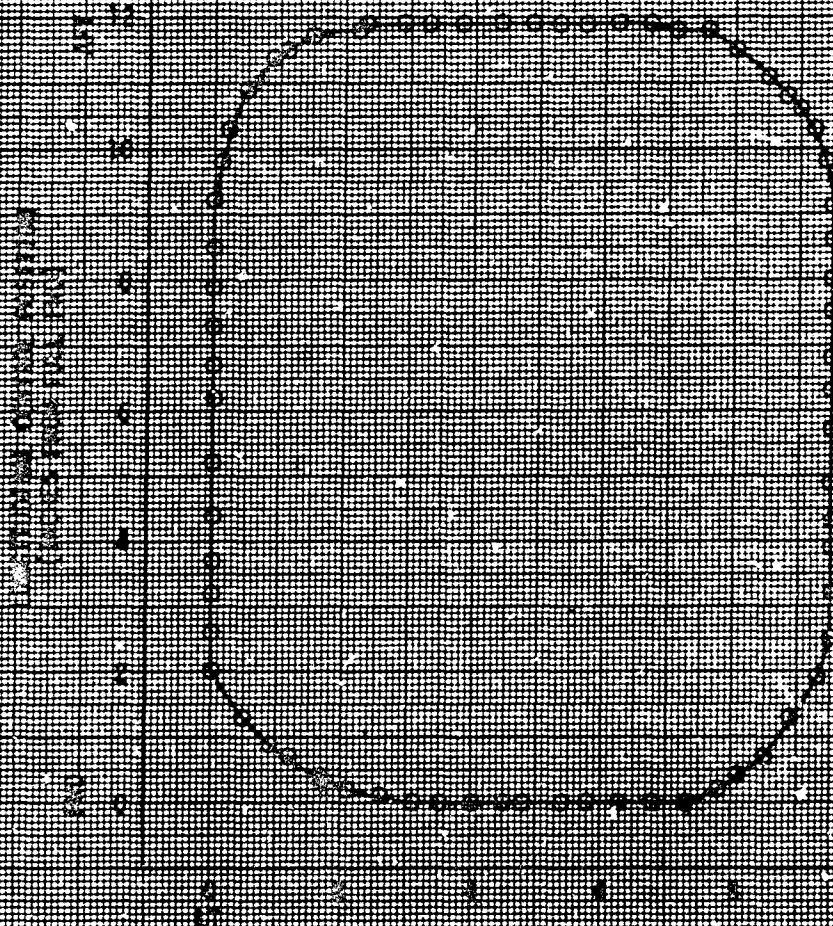
APPENDIX E. TEST DATA

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FORM 1 LIMITS OF CIRCLE CONTROL TRAIL OF THE U.S. AIR FORCE

- 1. PURPOSE: TO SHOW THE LIMITS OF CIRCLE CONTROL TRAIL.
- 2. SCOPE: THIS FORM IS APPLIED TO ALL CIRCLE CONTROL TRAILS.
- 3. REFERENCES: U.S. AIR FORCE REGULATION 10-1, 10-2, 10-3, 10-4, 10-5, 10-6, 10-7, 10-8, 10-9, 10-10, 10-11, 10-12, 10-13, 10-14, 10-15, 10-16, 10-17, 10-18, 10-19, 10-20, 10-21, 10-22, 10-23, 10-24, 10-25, 10-26, 10-27, 10-28, 10-29, 10-30, 10-31, 10-32, 10-33, 10-34, 10-35, 10-36, 10-37, 10-38, 10-39, 10-40, 10-41, 10-42, 10-43, 10-44, 10-45, 10-46, 10-47, 10-48, 10-49, 10-50, 10-51, 10-52, 10-53, 10-54, 10-55, 10-56, 10-57, 10-58, 10-59, 10-60, 10-61, 10-62, 10-63, 10-64, 10-65, 10-66, 10-67, 10-68, 10-69, 10-70, 10-71, 10-72, 10-73, 10-74, 10-75, 10-76, 10-77, 10-78, 10-79, 10-80, 10-81, 10-82, 10-83, 10-84, 10-85, 10-86, 10-87, 10-88, 10-89, 10-90, 10-91, 10-92, 10-93, 10-94, 10-95, 10-96, 10-97, 10-98, 10-99, 10-100.
- 4. DEFINITIONS: A circle control trail is a trail of circles of equal radius, the centers of which are located on a straight line.
- 5. LIMITS: The limits of a circle control trail are the circles of the trail which are tangent to the line of centers.
- 6. APPLICATION: This form is applied to all circle control trails.
- 7. LIMITS OF CIRCLE CONTROL TRAIL: The limits of a circle control trail are the circles of the trail which are tangent to the line of centers.



U.S. AIR FORCE
10-1, 10-2, 10-3, 10-4, 10-5, 10-6, 10-7, 10-8, 10-9, 10-10, 10-11, 10-12, 10-13, 10-14, 10-15, 10-16, 10-17, 10-18, 10-19, 10-20, 10-21, 10-22, 10-23, 10-24, 10-25, 10-26, 10-27, 10-28, 10-29, 10-30, 10-31, 10-32, 10-33, 10-34, 10-35, 10-36, 10-37, 10-38, 10-39, 10-40, 10-41, 10-42, 10-43, 10-44, 10-45, 10-46, 10-47, 10-48, 10-49, 10-50, 10-51, 10-52, 10-53, 10-54, 10-55, 10-56, 10-57, 10-58, 10-59, 10-60, 10-61, 10-62, 10-63, 10-64, 10-65, 10-66, 10-67, 10-68, 10-69, 10-70, 10-71, 10-72, 10-73, 10-74, 10-75, 10-76, 10-77, 10-78, 10-79, 10-80, 10-81, 10-82, 10-83, 10-84, 10-85, 10-86, 10-87, 10-88, 10-89, 10-90, 10-91, 10-92, 10-93, 10-94, 10-95, 10-96, 10-97, 10-98, 10-99, 10-100.

APPROXIMATE CONTROL SYSTEM CHARACTERISTICS FOR THE T-10

1. INITIAL STATE
2. INITIAL DISPLACEMENT FROM EQUILIBRIUM
3. INITIAL VELOCITY
4. INITIAL ACCELERATION
5. INITIAL POSITION
6. INITIAL VELOCITY
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93. INITIAL VELOCITY
94. INITIAL ACCELERATION
95. INITIAL POSITION
96. INITIAL VELOCITY
97. INITIAL ACCELERATION
98. INITIAL POSITION
99. INITIAL VELOCITY
100. INITIAL ACCELERATION

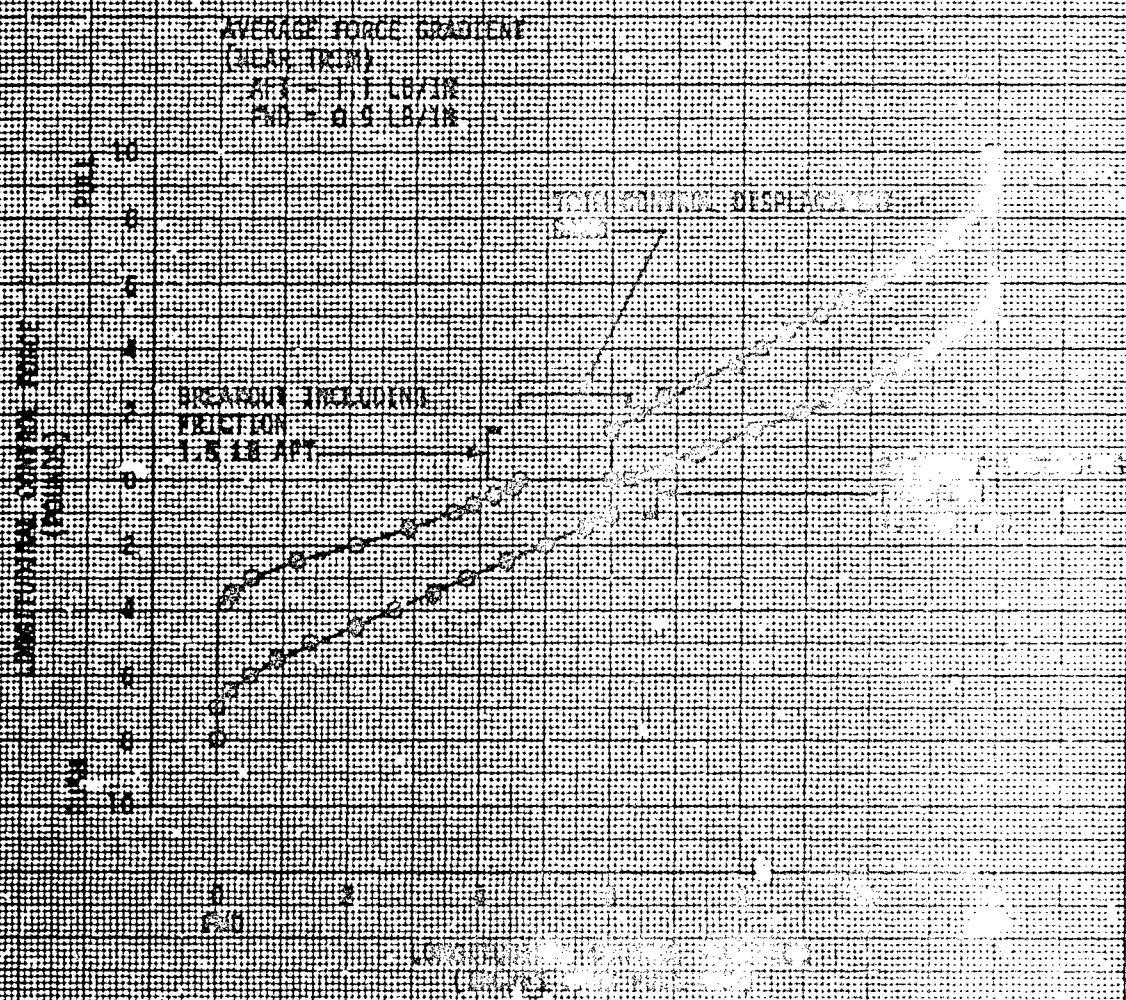


FIGURE 1 LATERAL CONTROL SYSTEM CHARACTERISTICS OF THE C-130A AIRCRAFT

- NOTES:
1. SYSTEM STATIC
 2. FORCES AND POSITIONS MEASURED AT CENTER OF GRAVITY
 3. LATERAL AND DIRECTIONAL FORCES PROVIDED BY DIRECTIONAL POWER UNIT
 4. TRIM FILTER ON
 5. TOTAL LATERAL CONTROL TRAVEL = 3.5 INCHES
 6. LONGITUDINAL CONTROL POSITION = 0.1 INCHES FROM FULL FORWARD
 7. FORCE TRIM ON, AIRCRAFT CYCLE FREQUENCY 0.75

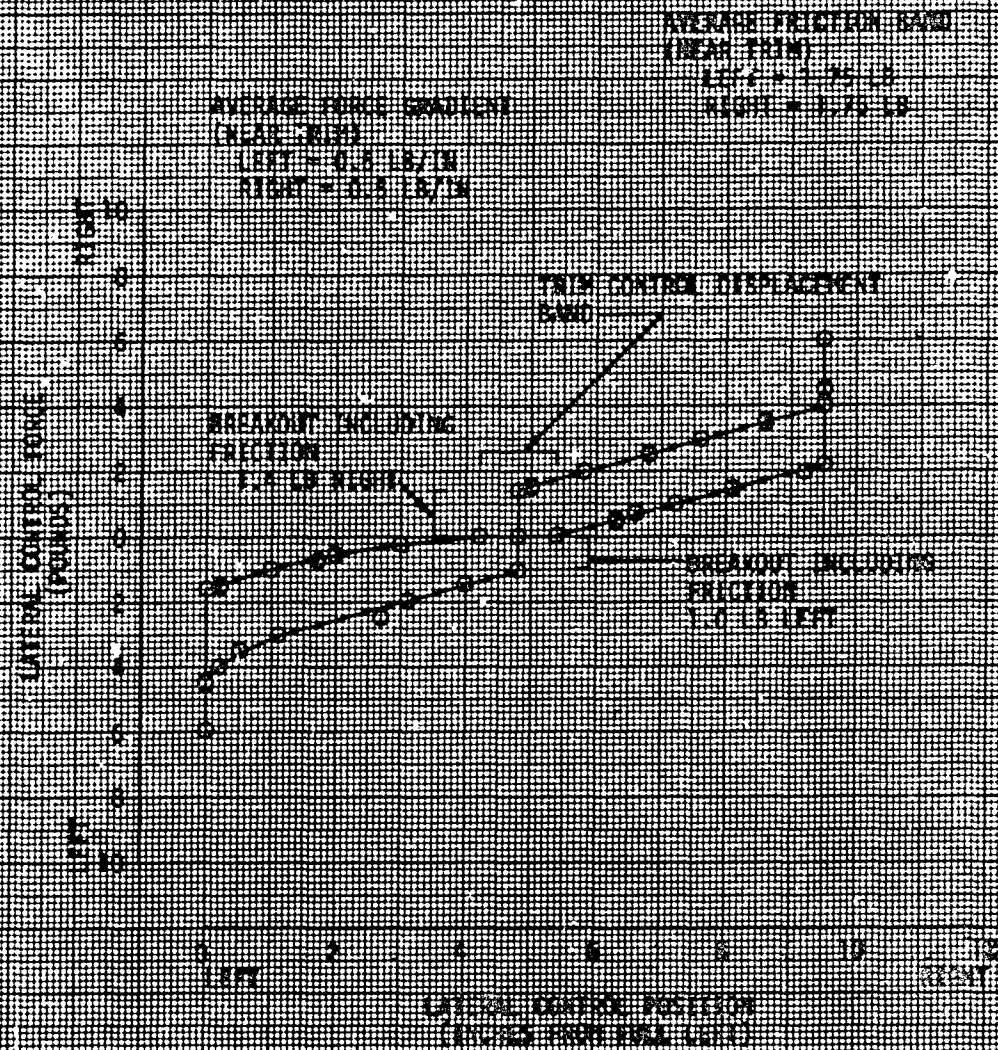
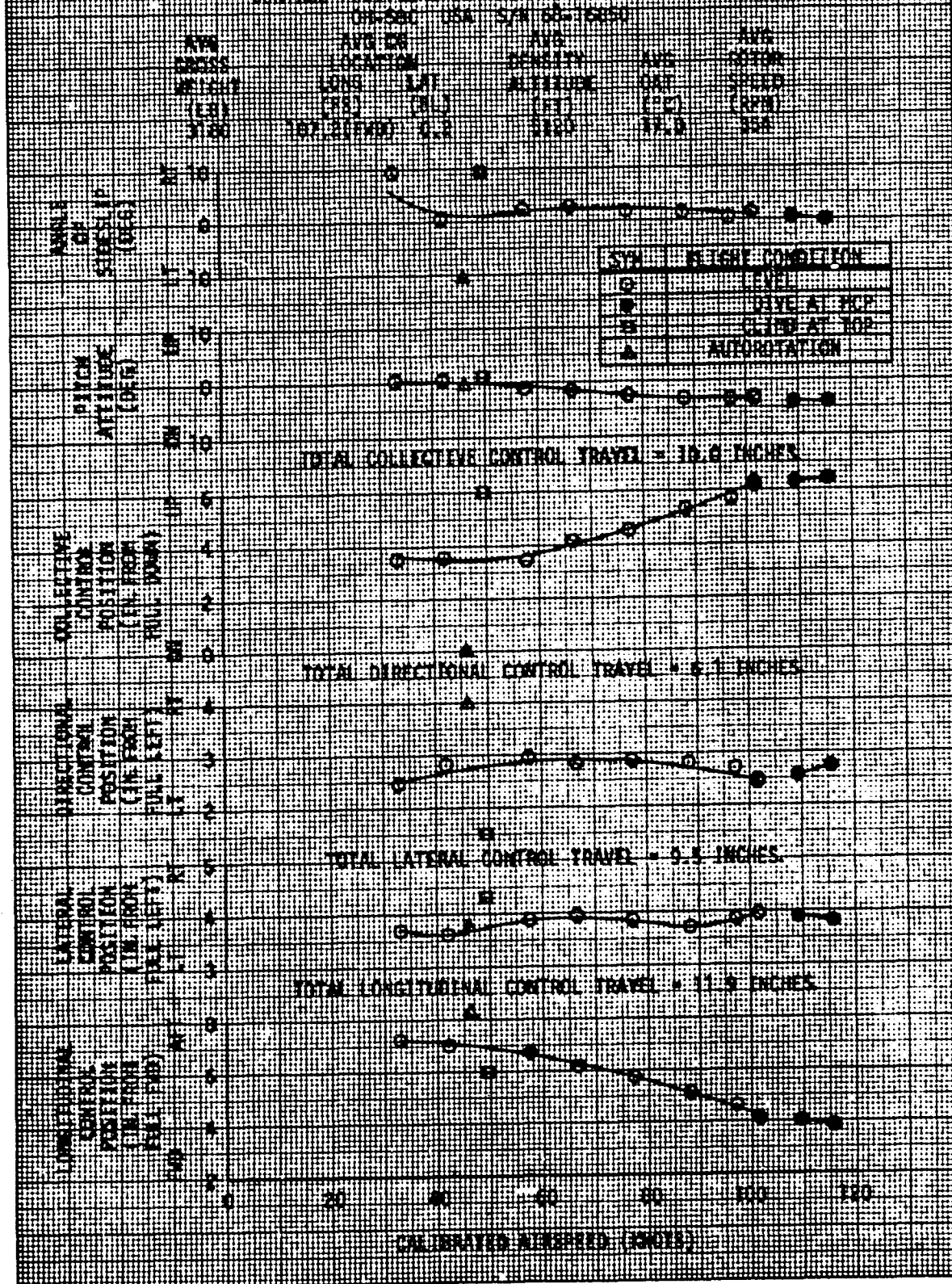
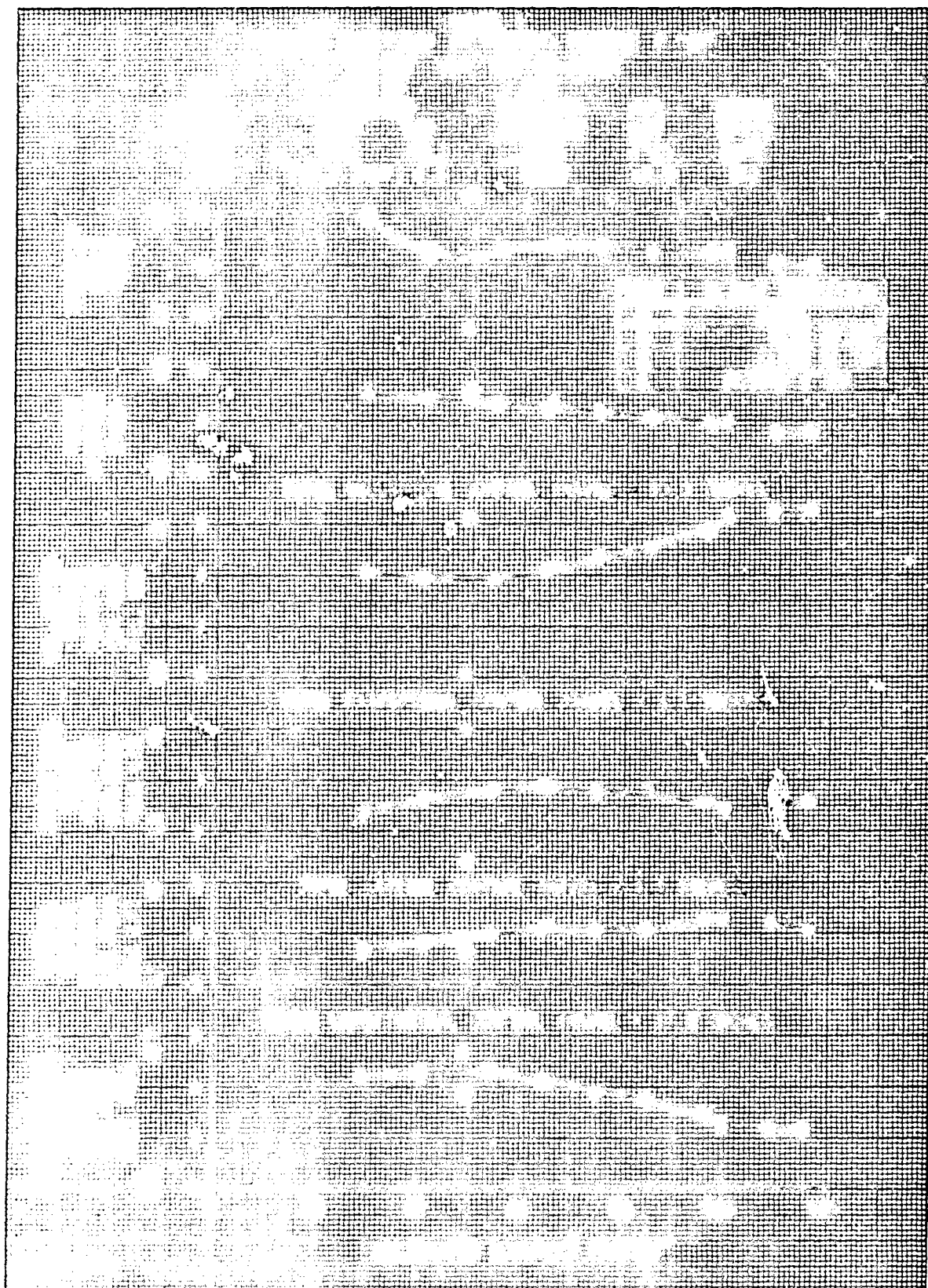
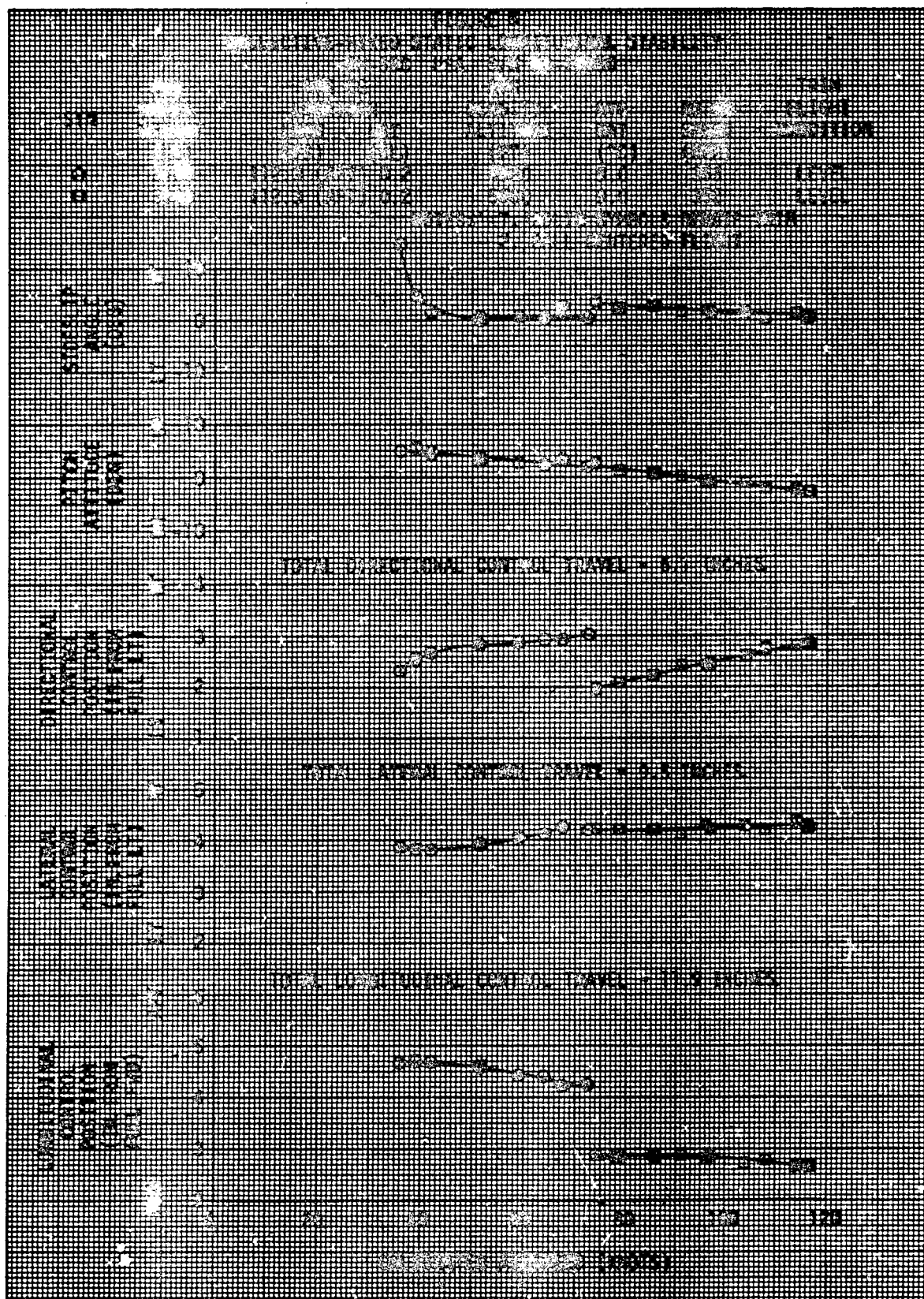
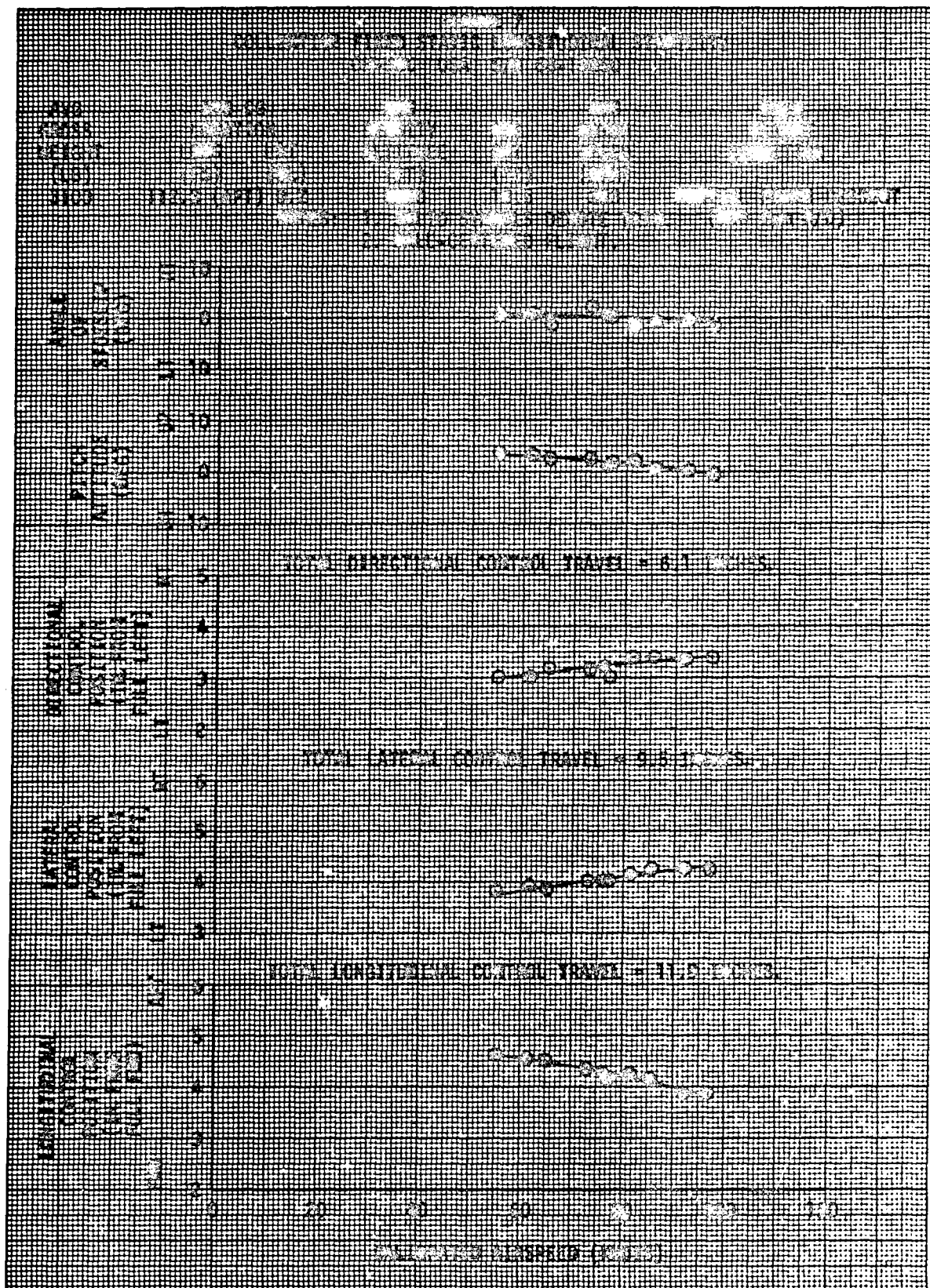


FIGURE 4
CONTROL POSITIONS IN TRAINED FORWARD FLIGHT









[illegible]

Number of hauls	<i>P. setiferus</i> (%)	<i>P. setiferus</i> + <i>P. setiferus</i> + <i>P. setiferus</i> (%)
1	10	5
2	30	10
3	50	15
4	70	18
5	85	20
6	95	22
7	100	23
8	100	24
9	100	25
10	100	26

1990	2	10
1991	2	10

Concentration of inhibitor	Rate of polymerization
0.0	1.0
0.1	0.9
0.2	0.8
0.3	0.7
0.4	0.6
0.5	0.5
0.6	0.4
0.7	0.3
0.8	0.2
0.9	0.1
1.0	0.0

Year	Number of cases (thousands)
1990	1.5
1991	11.5
1992	0.5
1993	1.0
1994	1.5
1995	10.5
1996	1.0
1997	1.2
1998	1.3
1999	1.4
2000	1.5

REF ID: A66544

100-443887-100

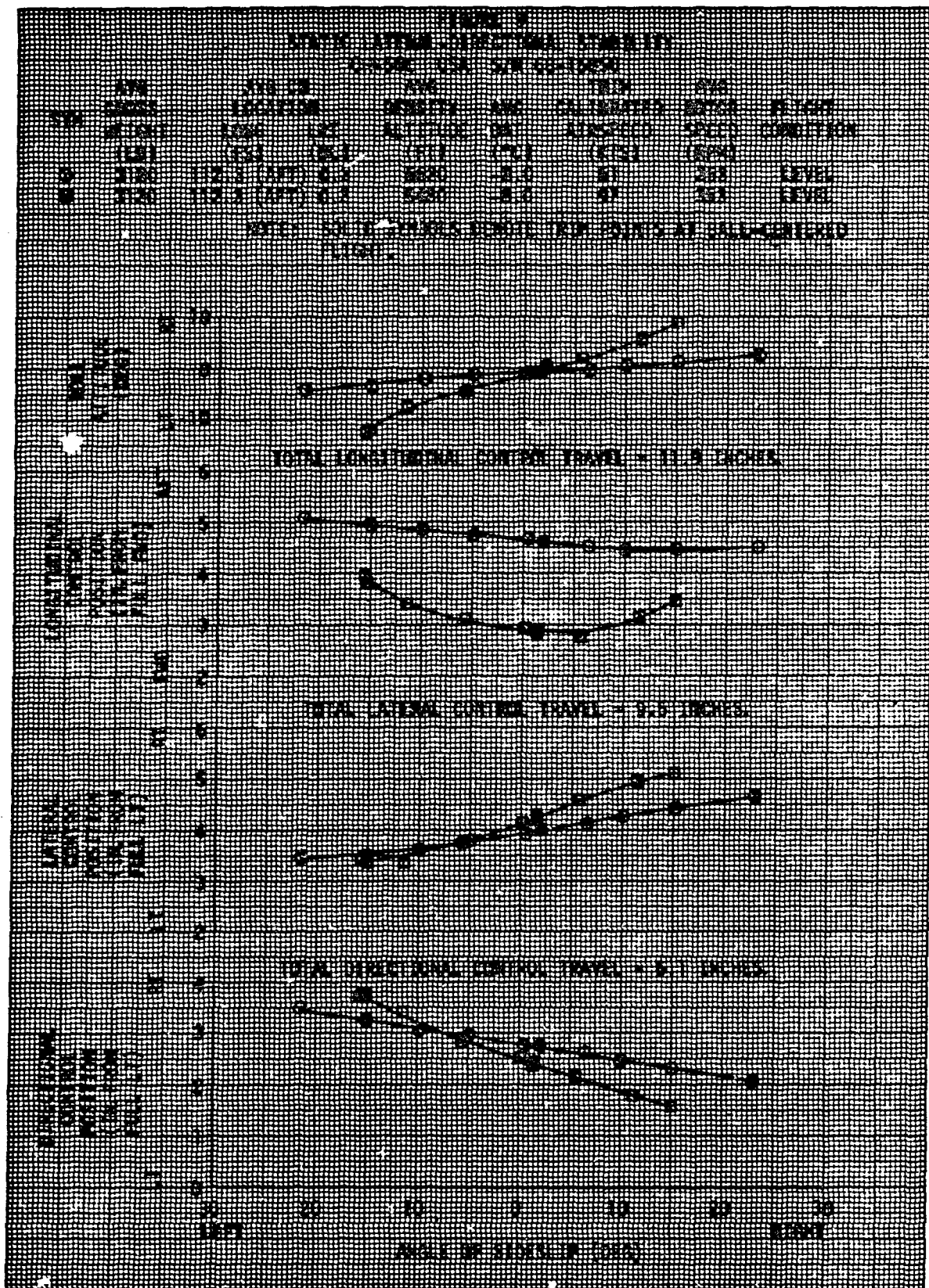
Figure 1 is a line graph titled "Percentage of total population in the labor force by age group, 1960-1990". The vertical axis (Y-axis) is labeled "Percentage of total population in the labor force" and ranges from 0 to 100 in increments of 10. The horizontal axis (X-axis) is labeled "Age" and ranges from 0 to 60 in increments of 10. There are five data series representing different age groups: 15-24, 25-34, 35-44, 45-54, and 55-60. All series show a consistent downward trend from 1960 to 1990. The 15-24 age group starts at approximately 65% in 1960 and ends at about 45% in 1990. The 25-34 age group starts at about 55% and ends at 35%. The 35-44 age group starts at about 45% and ends at 25%. The 45-54 age group starts at about 35% and ends at 15%. The 55-60 age group starts at about 25% and ends at 10%.

Age Group	1960 (%)	1970 (%)	1980 (%)	1990 (%)
15-24	65	60	55	45
25-34	55	50	45	35
35-44	45	40	35	25
45-54	35	30	25	15
55-60	25	20	15	10

C	R _p
0.0	1.00
0.1	0.95
0.2	0.90
0.3	0.85
0.4	0.80
0.5	0.75
0.6	0.70
0.7	0.65
0.8	0.60
0.9	0.55
1.0	0.50

100

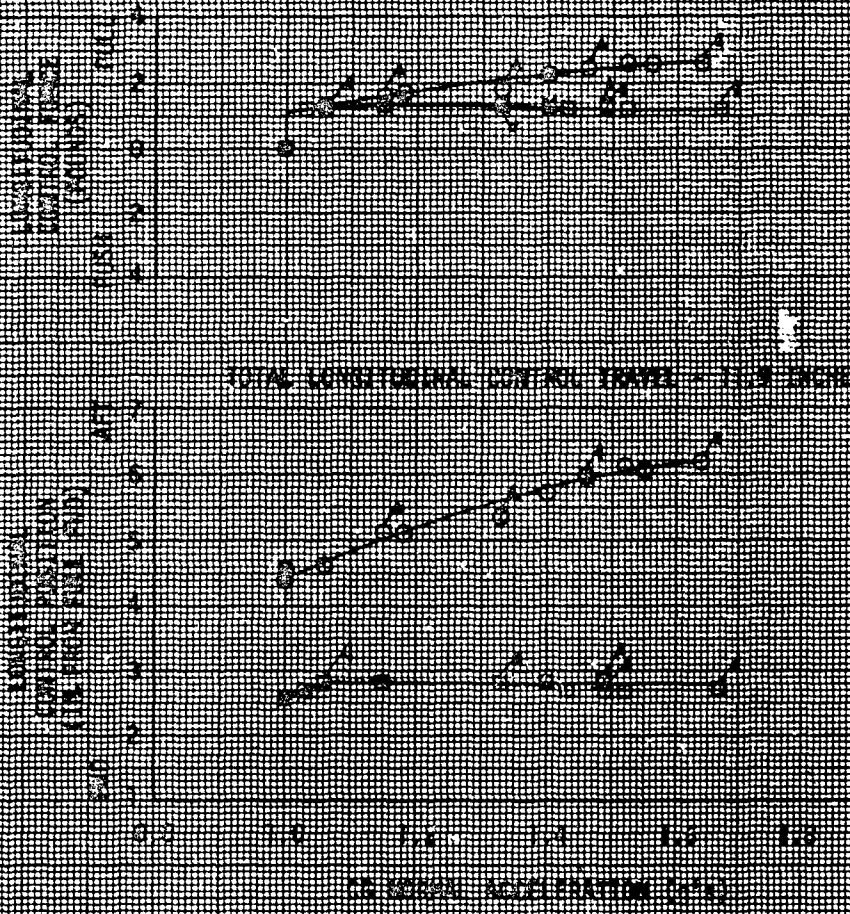
THE UNIVERSITY OF CHICAGO



PROBABILITY STUDY 1952-53

DATE	TIME	LOCATION	WIND	TEMP	REL	WIND	TEMP	REL
10/10	10:00	100	10	10	10	10	10	10
10/11	10:00	100	10	10	10	10	10	10
10/12	10:00	100	10	10	10	10	10	10
10/13	10:00	100	10	10	10	10	10	10
10/14	10:00	100	10	10	10	10	10	10
10/15	10:00	100	10	10	10	10	10	10
10/16	10:00	100	10	10	10	10	10	10
10/17	10:00	100	10	10	10	10	10	10
10/18	10:00	100	10	10	10	10	10	10
10/19	10:00	100	10	10	10	10	10	10
10/20	10:00	100	10	10	10	10	10	10
10/21	10:00	100	10	10	10	10	10	10
10/22	10:00	100	10	10	10	10	10	10
10/23	10:00	100	10	10	10	10	10	10
10/24	10:00	100	10	10	10	10	10	10
10/25	10:00	100	10	10	10	10	10	10
10/26	10:00	100	10	10	10	10	10	10
10/27	10:00	100	10	10	10	10	10	10
10/28	10:00	100	10	10	10	10	10	10
10/29	10:00	100	10	10	10	10	10	10
10/30	10:00	100	10	10	10	10	10	10
10/31	10:00	100	10	10	10	10	10	10

1. PROBABILITY STUDY
2. PROBABILITY STUDY
3. PROBABILITY STUDY
4. PROBABILITY STUDY
5. PROBABILITY STUDY
6. PROBABILITY STUDY
7. PROBABILITY STUDY
8. PROBABILITY STUDY
9. PROBABILITY STUDY
10. PROBABILITY STUDY



TIME	AVG	MAX	MIN	STDEV	COEFF	VAR	STDEV
0.0	0.00	0.00	0.00	0.00	0.00	0.00	0.00
0.1	0.00	0.00	0.00	0.00	0.00	0.00	0.00
0.2	0.00	0.00	0.00	0.00	0.00	0.00	0.00
0.3	0.00	0.00	0.00	0.00	0.00	0.00	0.00

1. ALL CONTROL TRACES
2. CONTROL TRACES EXCEPT LEFT TRACES
3. CONTROL TRACES EXCEPT RIGHT TRACES
4. CONTROL TRACES EXCEPT BOTH LEFT AND RIGHT TRACES
5. CONTROL TRACES EXCEPT DATA OBTAINED FROM

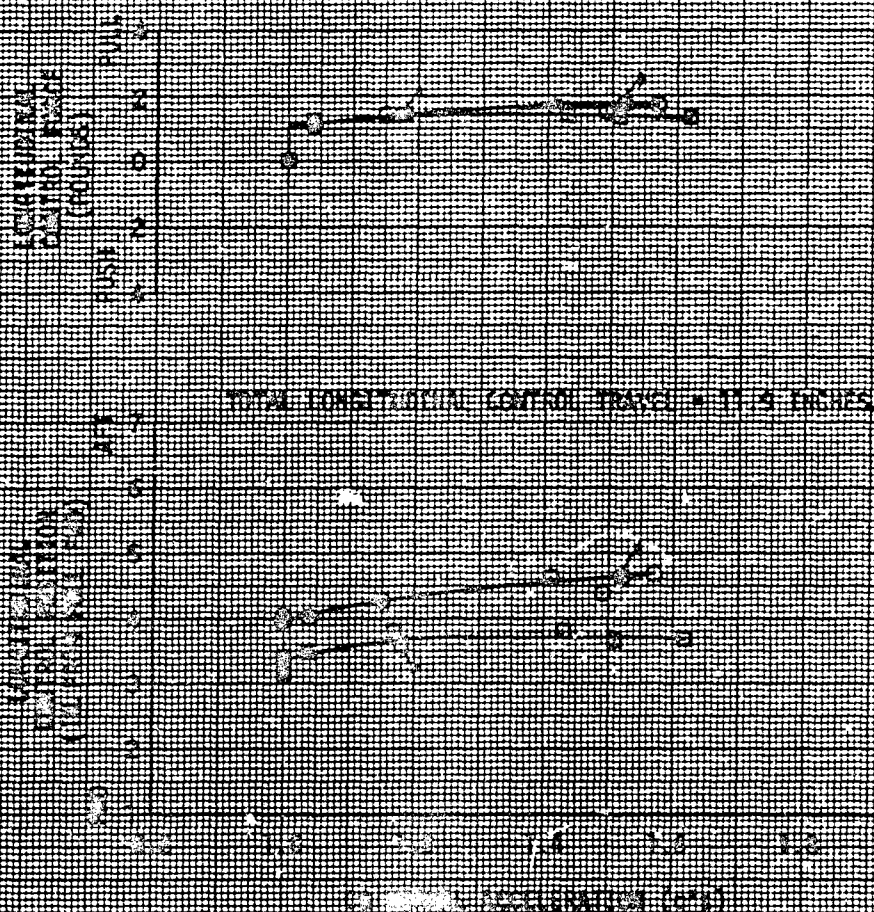


FIGURE 12
LONGITUDINAL LONG TERM RESPONSE
OH-580 USA S/N 63-15850

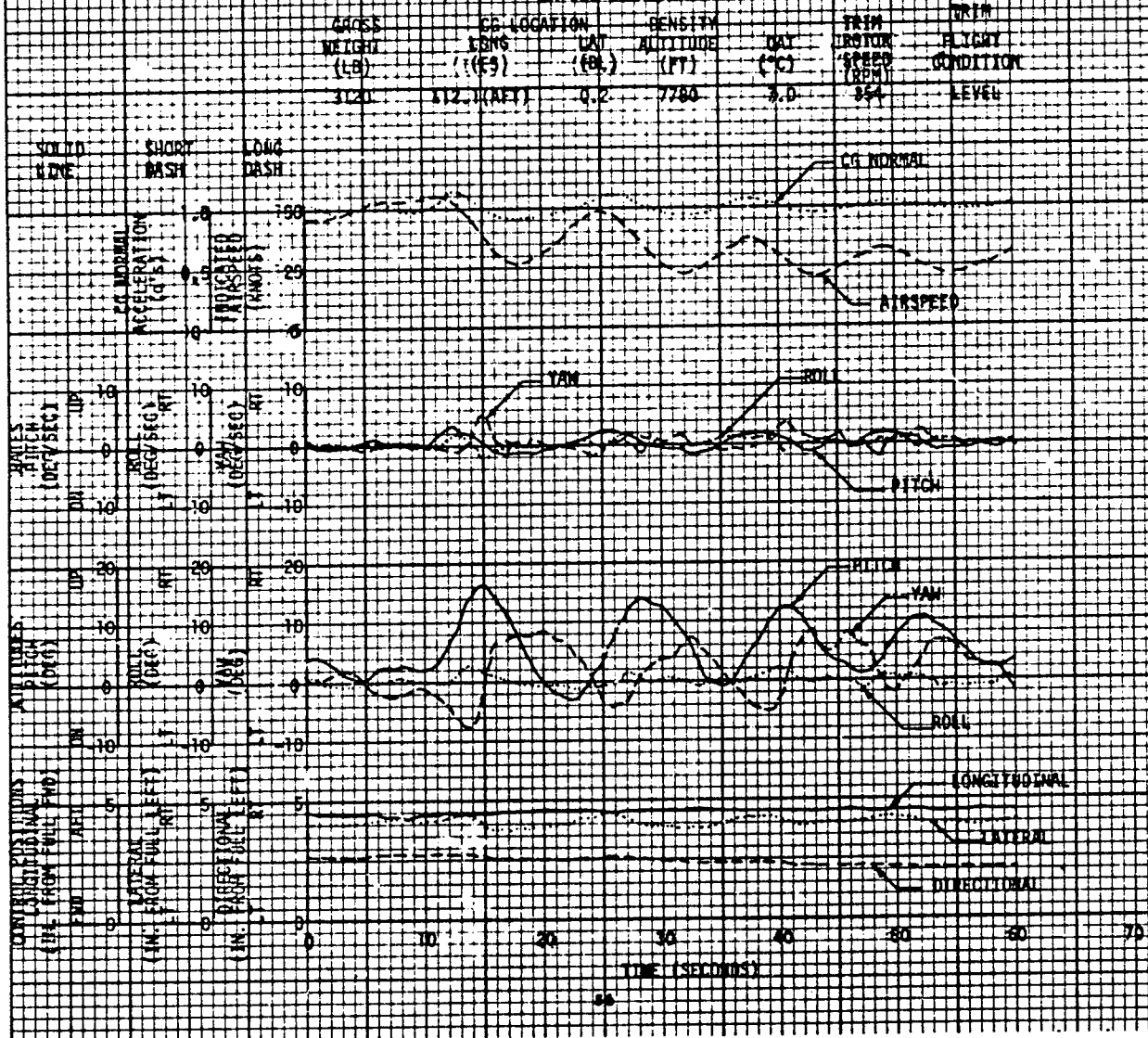
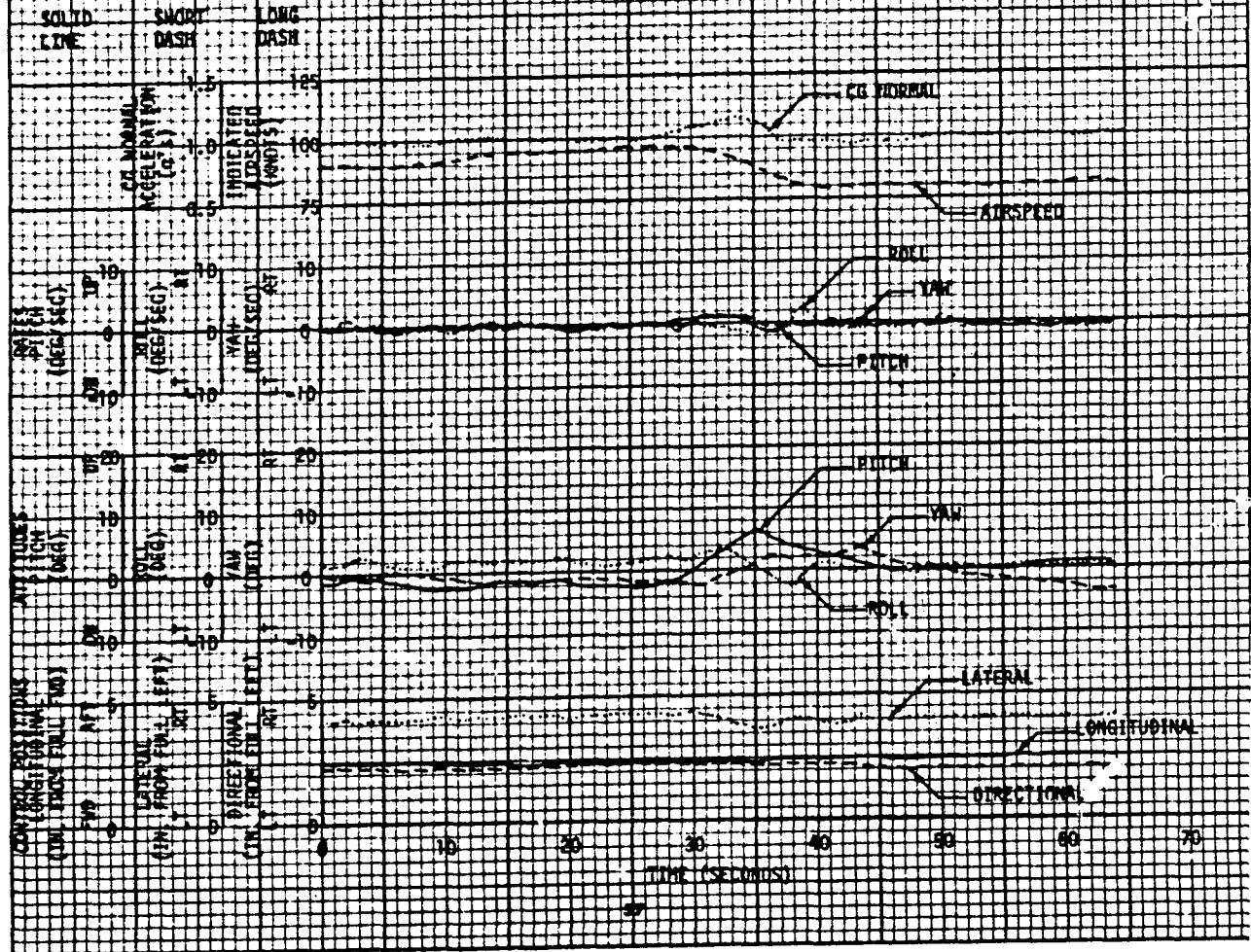


FIGURE 13

LONGITUDINAL LONG TERM RESPONSE
DH-60 USA S/N 48-16860

GRASS WEIGHT	LG LOCATION (FEET)	DENSITY ALTITUDE (FT)	OAT (°C)	TRIM MOTOR SPEED (RPM)	TRIM FLIGHT CONDITION
1080	112.0 (AFT)	0.2	7.5	854	LEVEL



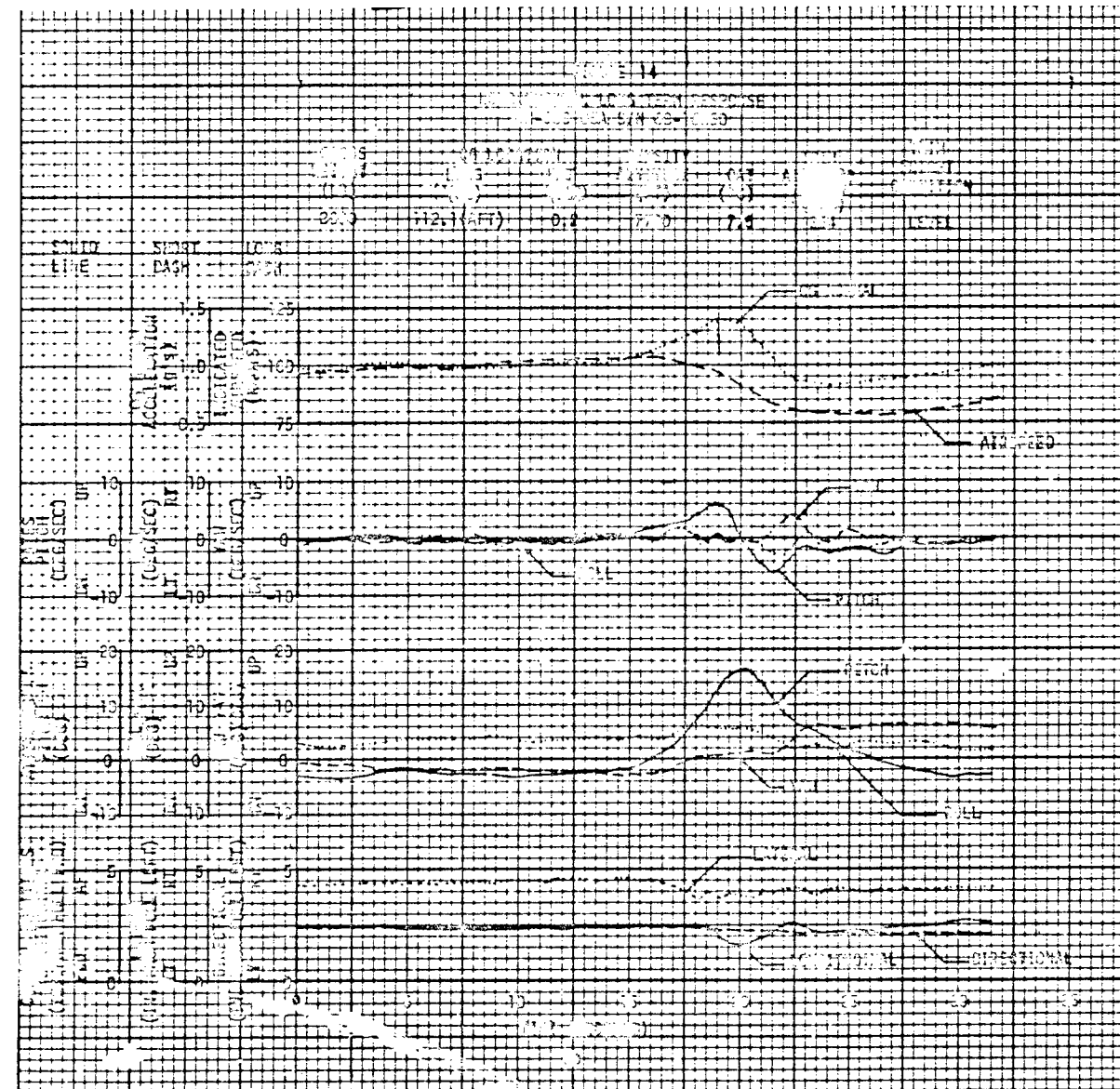


FIGURE 15
RELEASE FROM STEADY SIDESLIP
ON EBC USA 3/11/58-16850

CROSS WEIGHT (LB)	OR LOCATION (FEET)	SENSITIVITY (IN)	ALTITUDE (FT)	WIND SPEED (KPH)	WIND ALTITUDE (FEET)
3180	112.1 (AFT)	0.2	7300	1.0	354

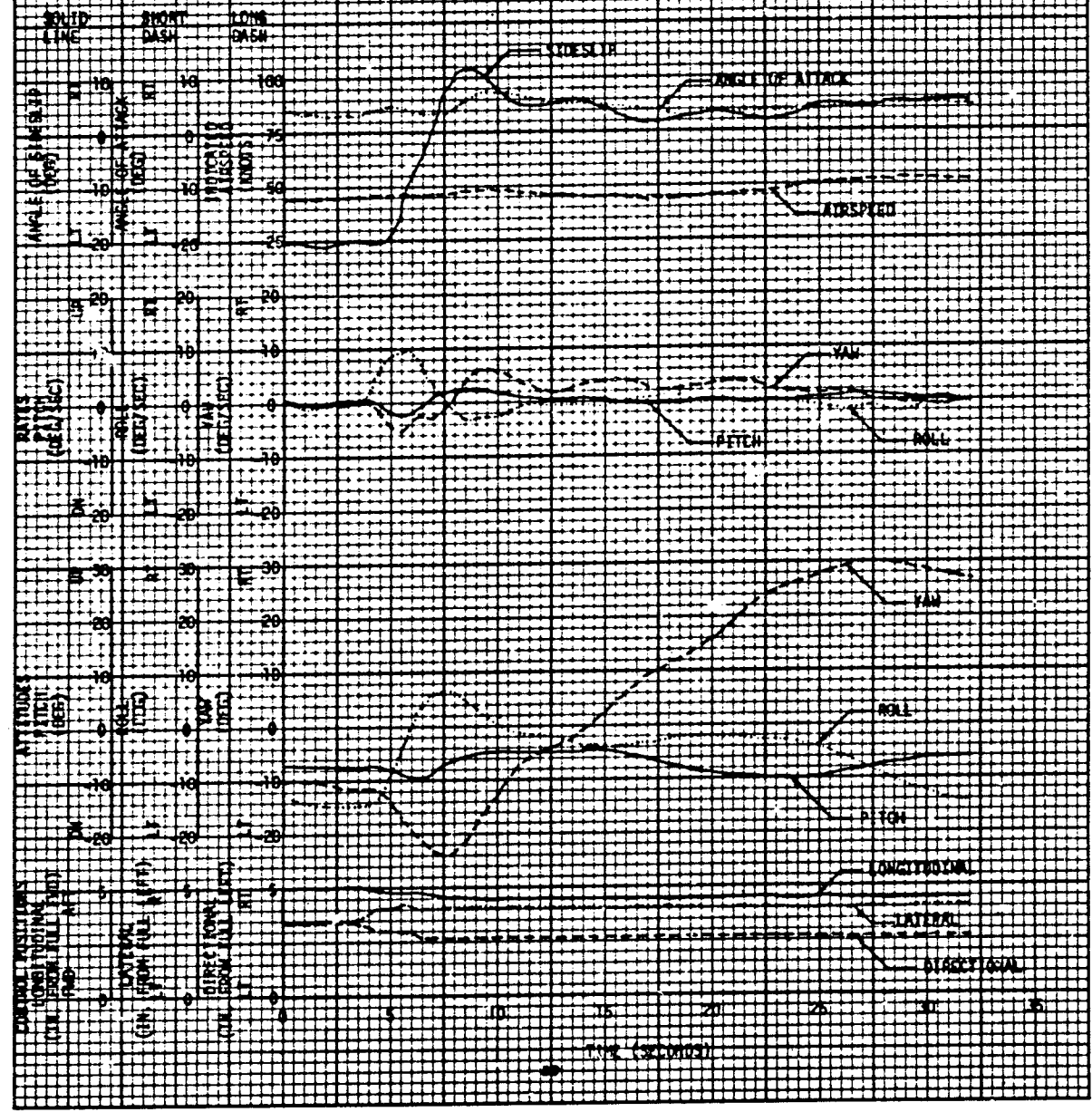


FIGURE 10
RELEASE FROM STEADY SIDESLIP
DN-38C USA S/N 68-16850

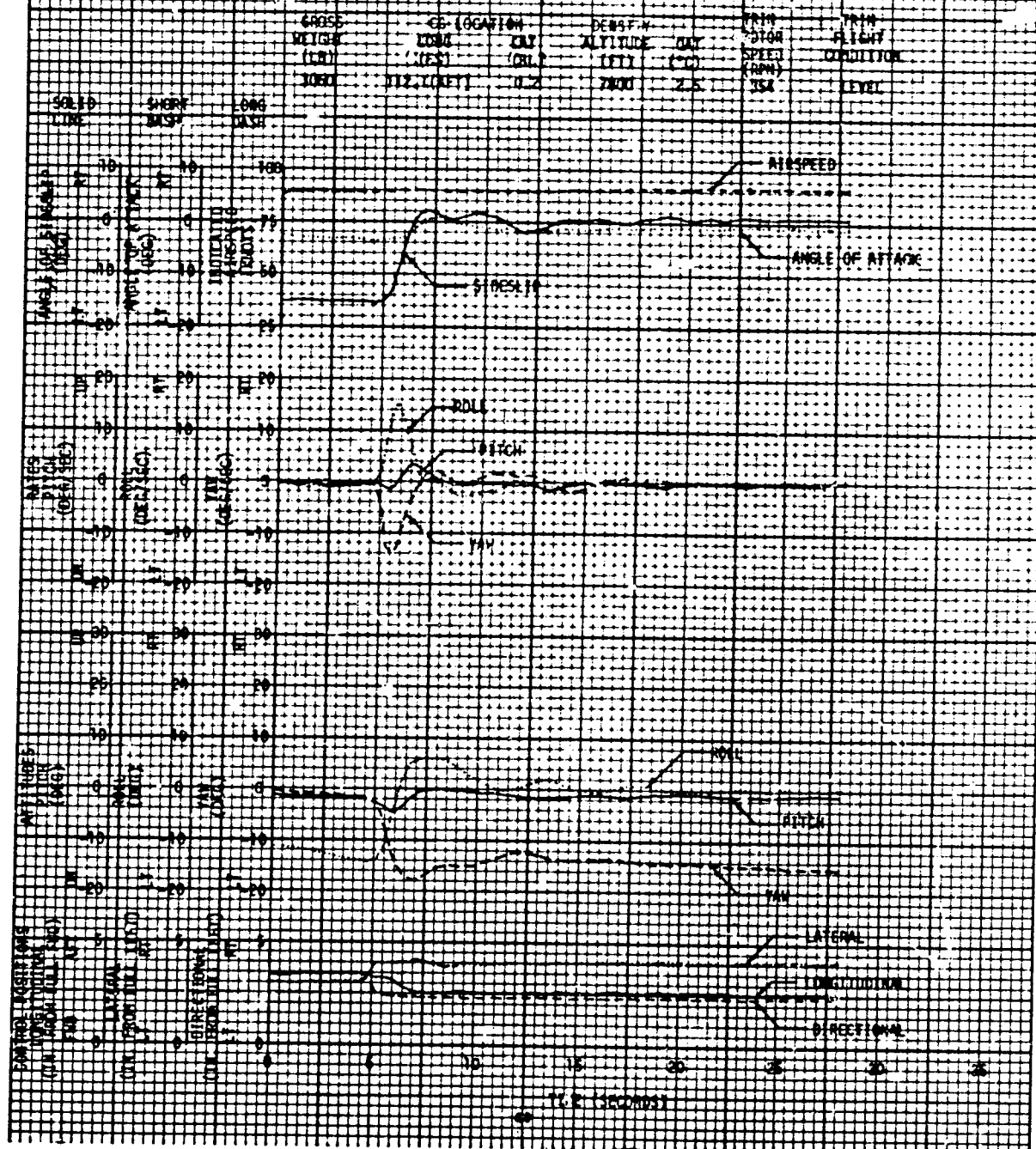


FIGURE 17
SPINAL STABILITY
ON: CH. 15A 57A 100-1000

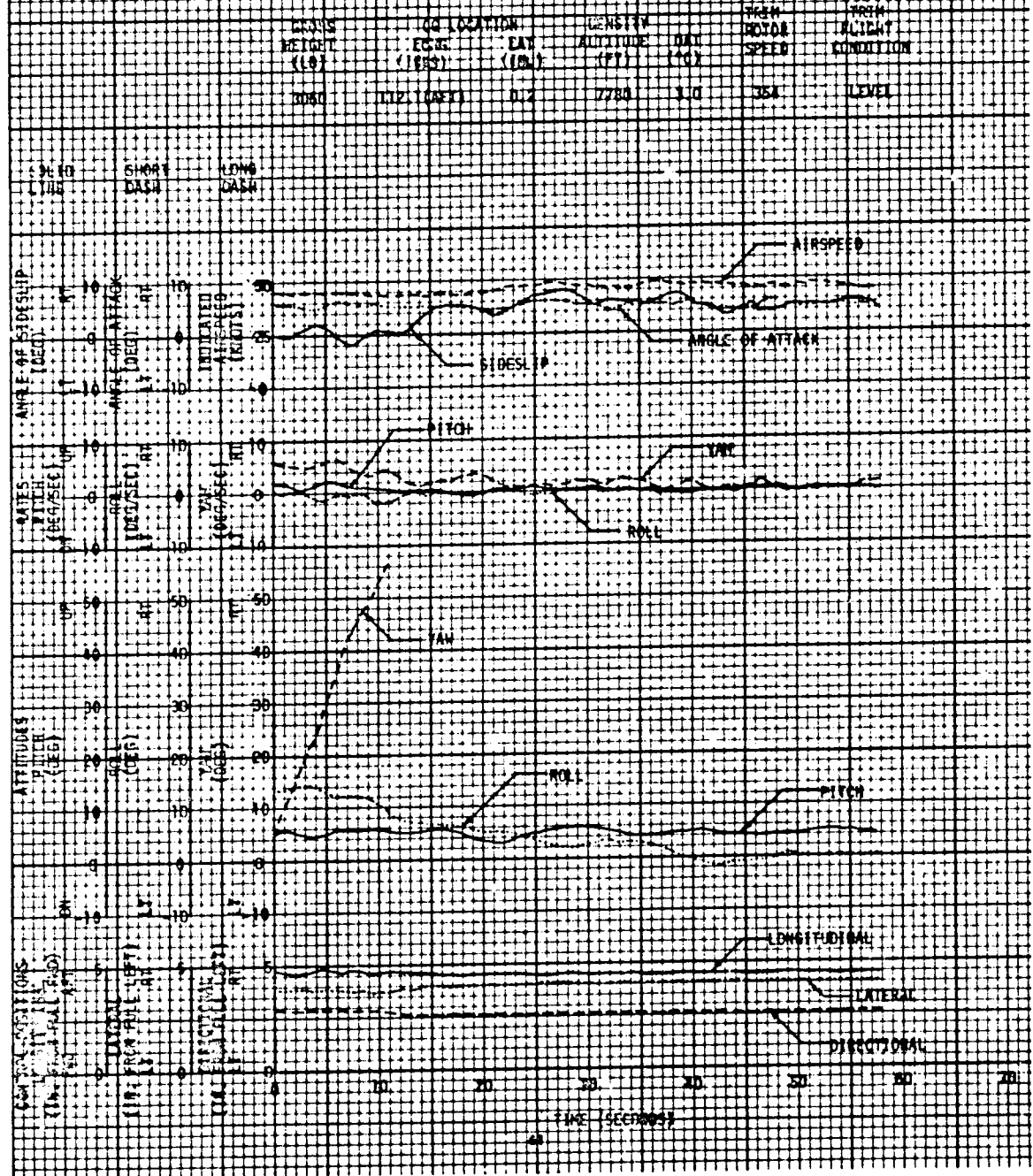
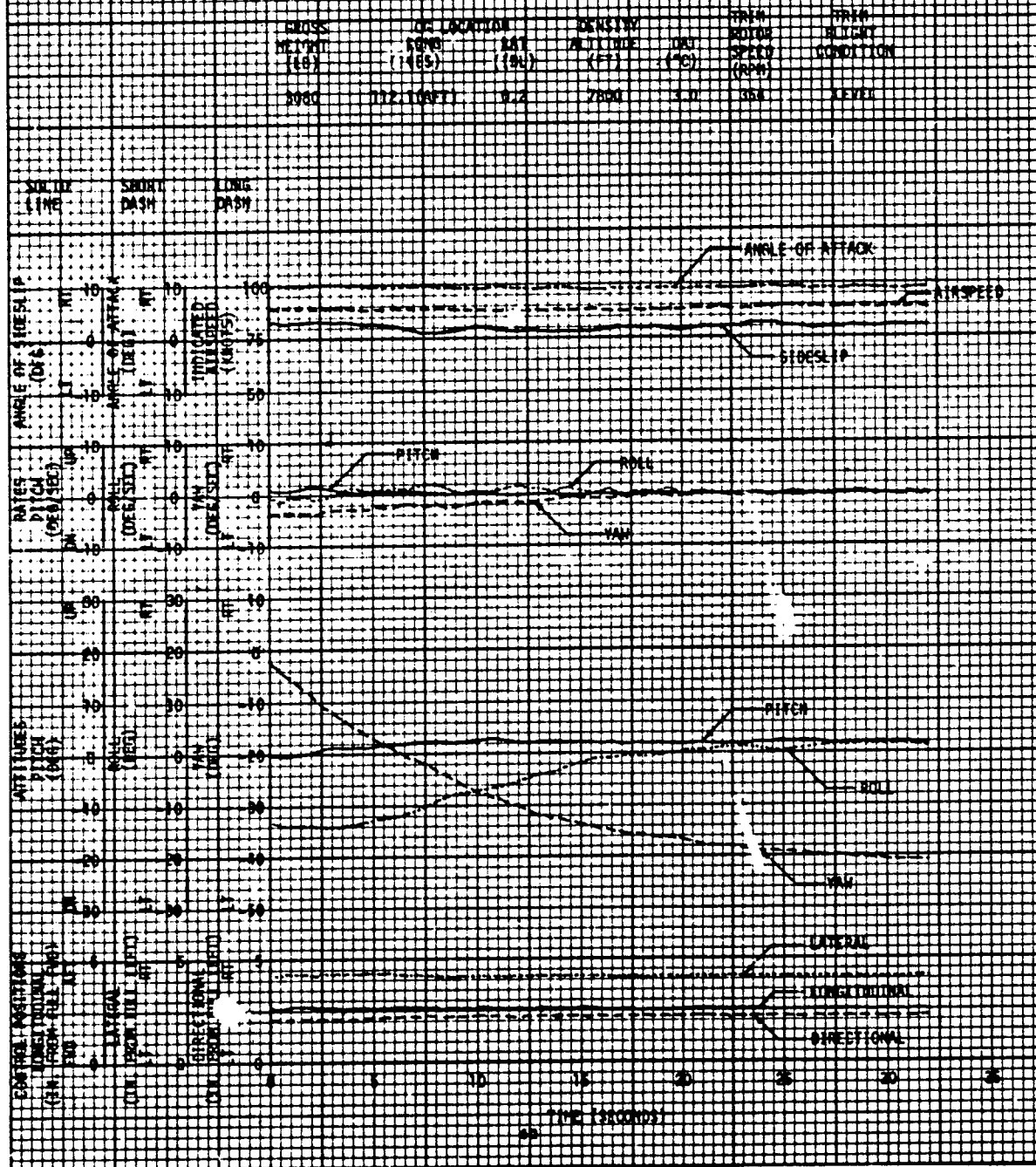
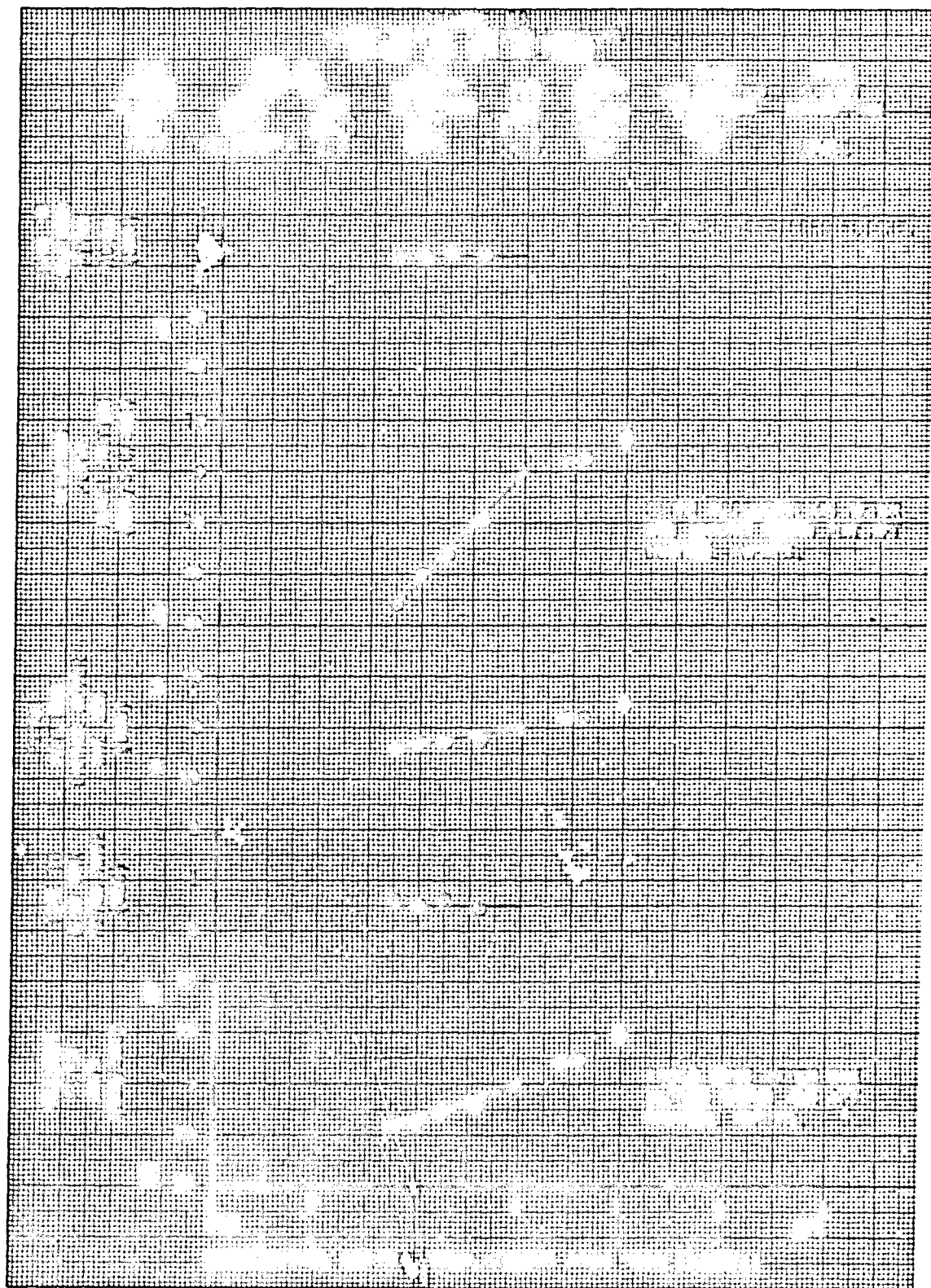
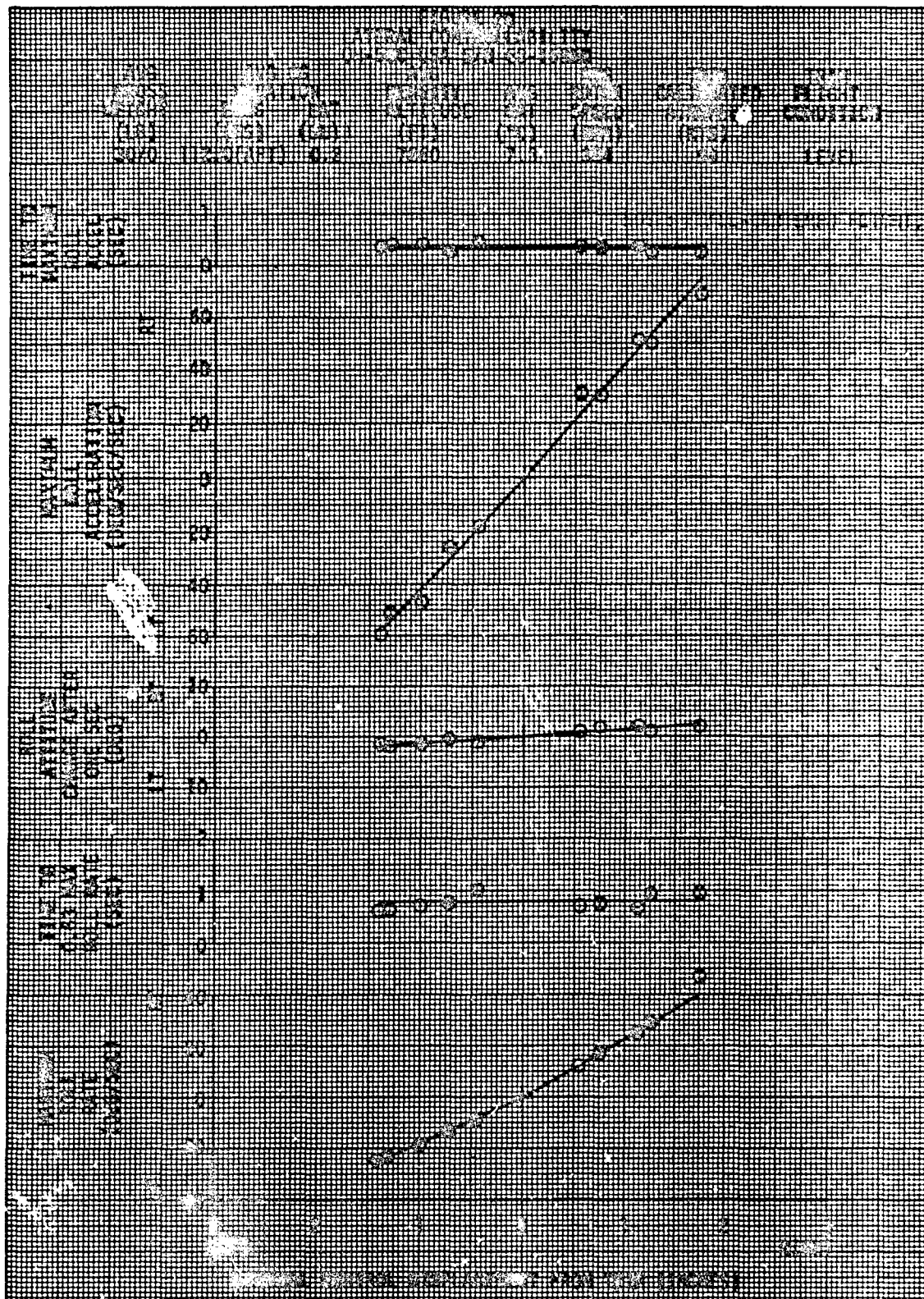
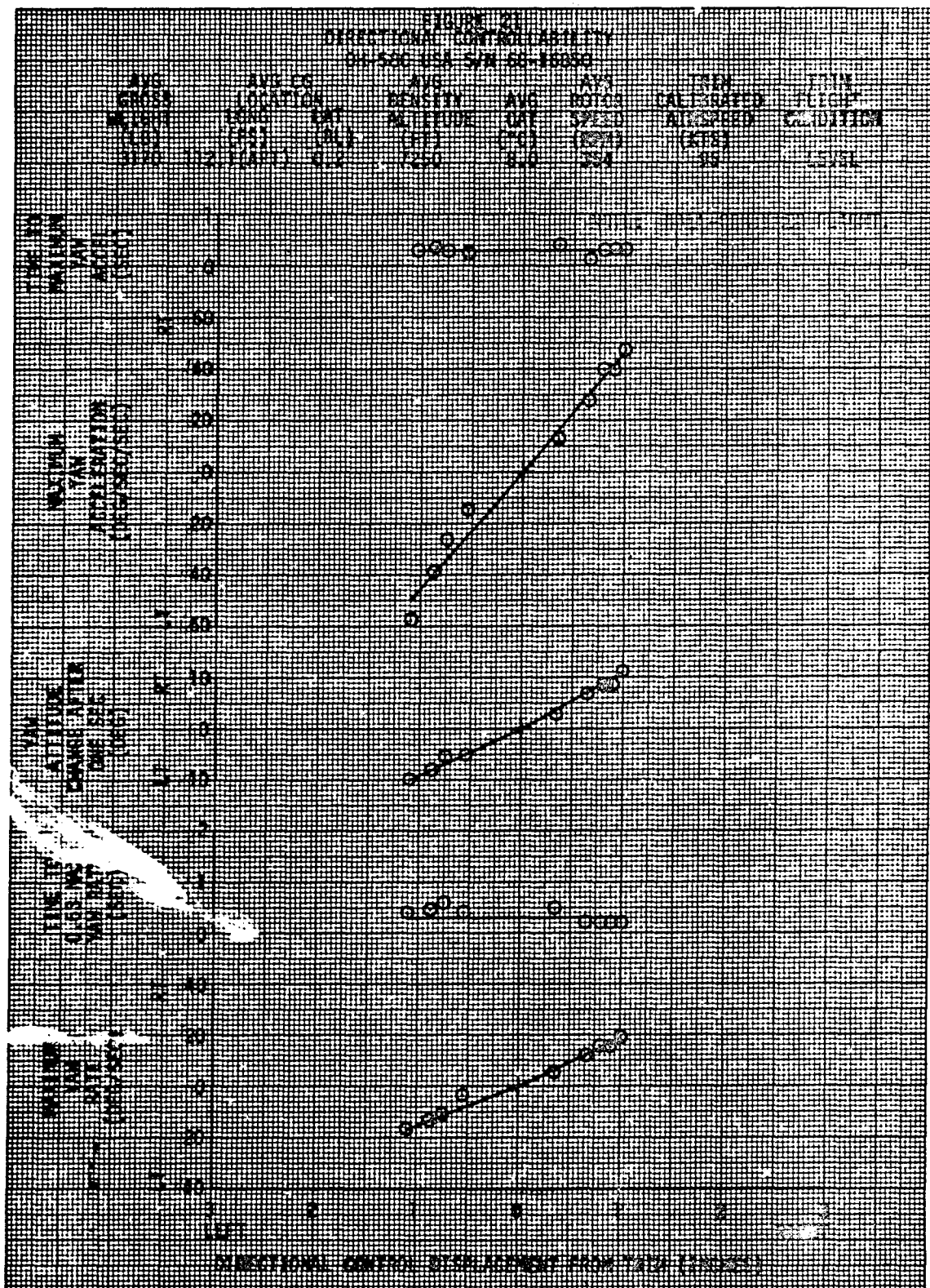


FIGURE 18
SPIRAL STABILITY
W-50C, CSM 574 DEL-16850









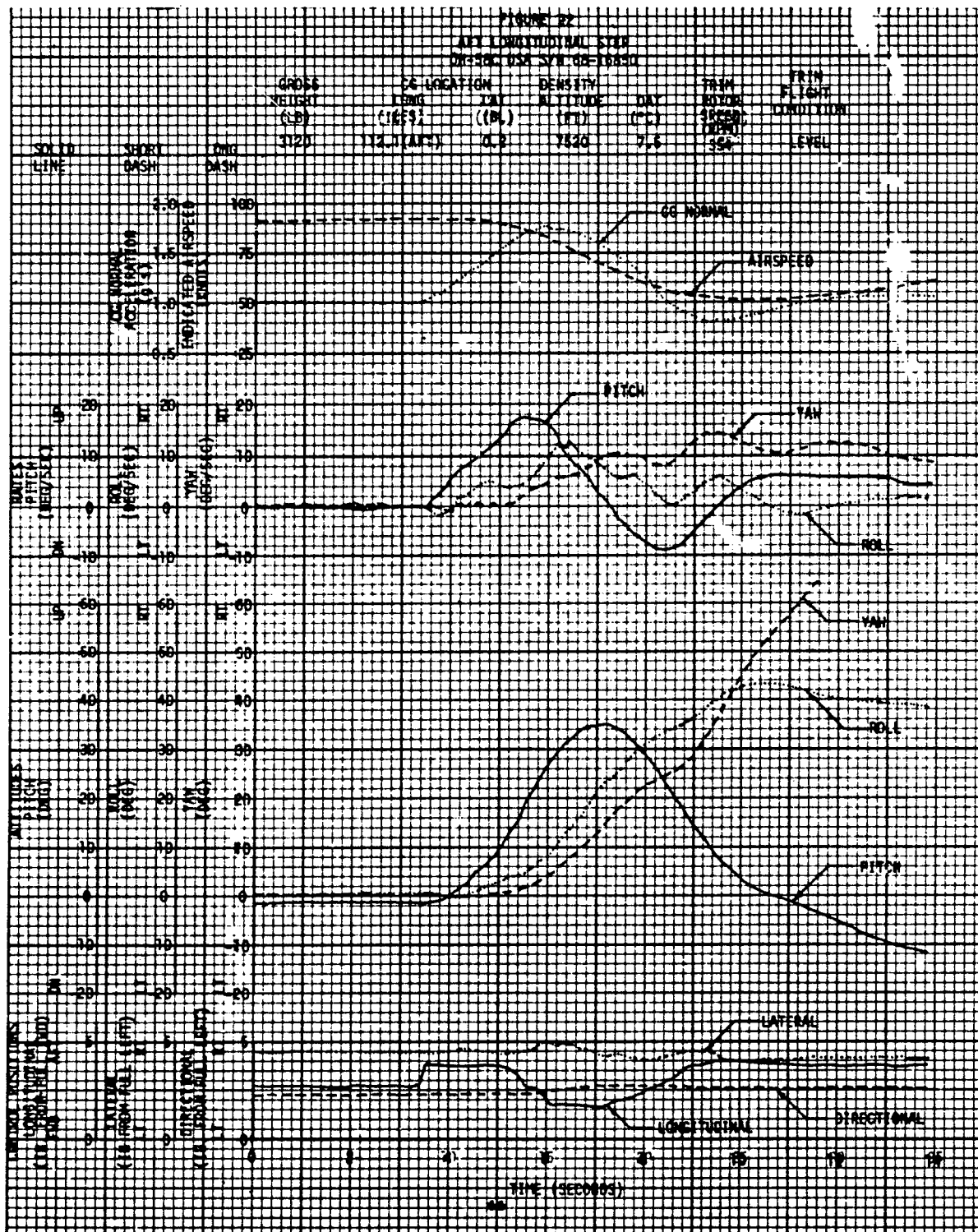


FIGURE 23
 LONGITUDINAL CONTROL AND LATERAL CONTROL
 ATTEMPTED STABILIZED POINT

TIME (SEC)	LONGITUDINAL CONTROL POSITION (IN)	LONGITUDINAL CONTROL POSITION (IN)	LONGITUDINAL CONTROL POSITION (IN)	LONGITUDINAL CONTROL POSITION (IN)	LONGITUDINAL CONTROL POSITION (IN)	LONGITUDINAL CONTROL POSITION (IN)
0.0	0.0	0.0	0.0	0.0	0.0	0.0
1.0	0.0	0.0	0.0	0.0	0.0	0.0
2.0	0.0	0.0	0.0	0.0	0.0	0.0
3.0	0.0	0.0	0.0	0.0	0.0	0.0
4.0	0.0	0.0	0.0	0.0	0.0	0.0
5.0	0.0	0.0	0.0	0.0	0.0	0.0
6.0	0.0	0.0	0.0	0.0	0.0	0.0
7.0	0.0	0.0	0.0	0.0	0.0	0.0
8.0	0.0	0.0	0.0	0.0	0.0	0.0
9.0	0.0	0.0	0.0	0.0	0.0	0.0
10.0	0.0	0.0	0.0	0.0	0.0	0.0

NOTE: 1. LONGITUDINAL CONTROL POSITION DURING
 ATTEMPTED STABILIZED POINT

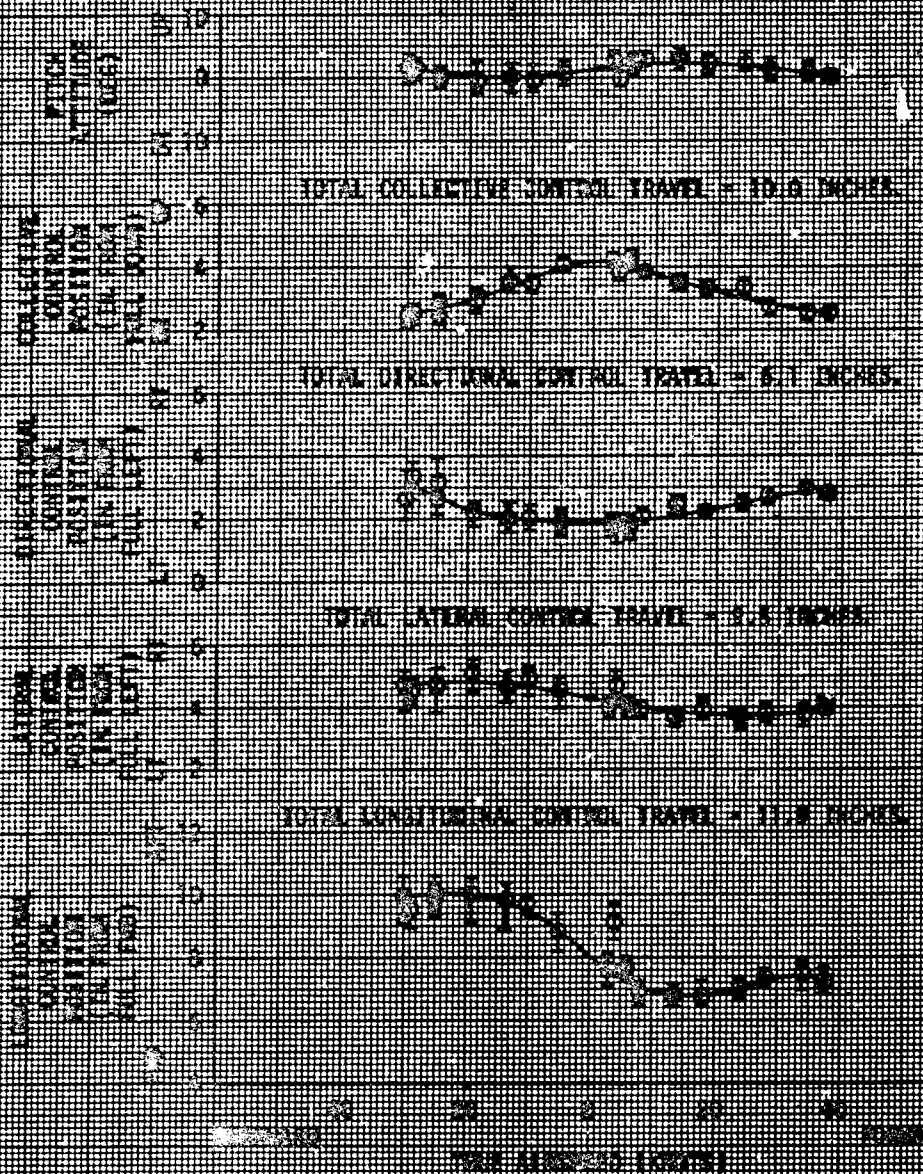


FIGURE 24 SUMMARY OF DATA ON THE 100' X 10' X 10' TEST

AVG GROSS WEIGHT (LB)	AVG NET WEIGHT (LB)	AVG NET WEIGHT (LB)	AVG NET WEIGHT (LB)	AVG NET WEIGHT (LB)	AVG NET WEIGHT (LB)
100	100	100	100	100	100

NOTE: ALL DATA ARE BASED ON THE 100' X 10' X 10' TEST.

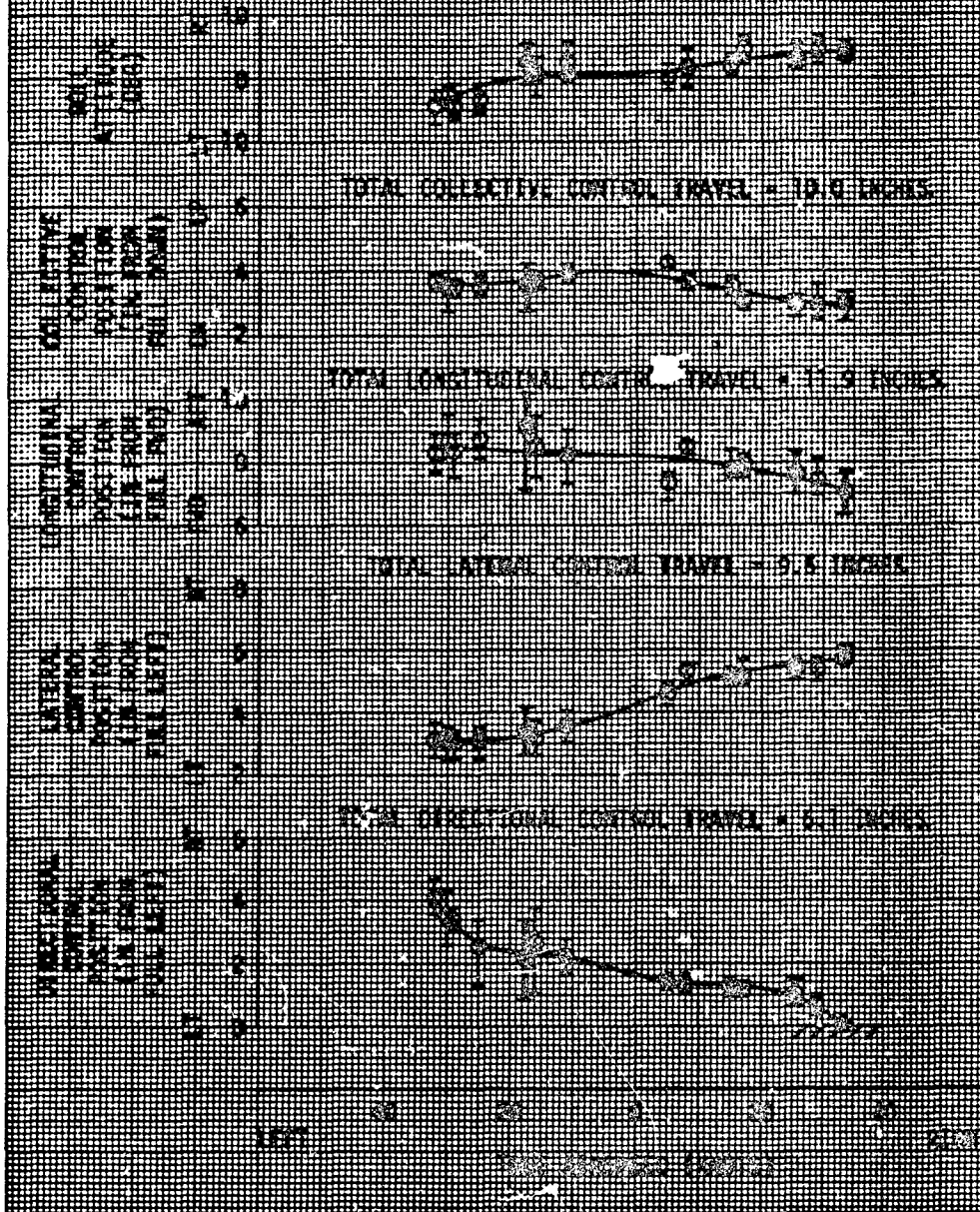


FIGURE 25

SIMULATED SUDDEN ENGINE FAILURE
ON-540 USA 3/N 88-16880

GROSS WEIGHT (LB)	CG LOCATION (IN)	DENSITY ALTITUDE (FT)	QAT (G)	TRIM AIRSPEED (KIAS)	INITIAL FLIGHT CONDITION
3110	109.4 (F40)	6.2	5940	17.0	95
					LEVEL

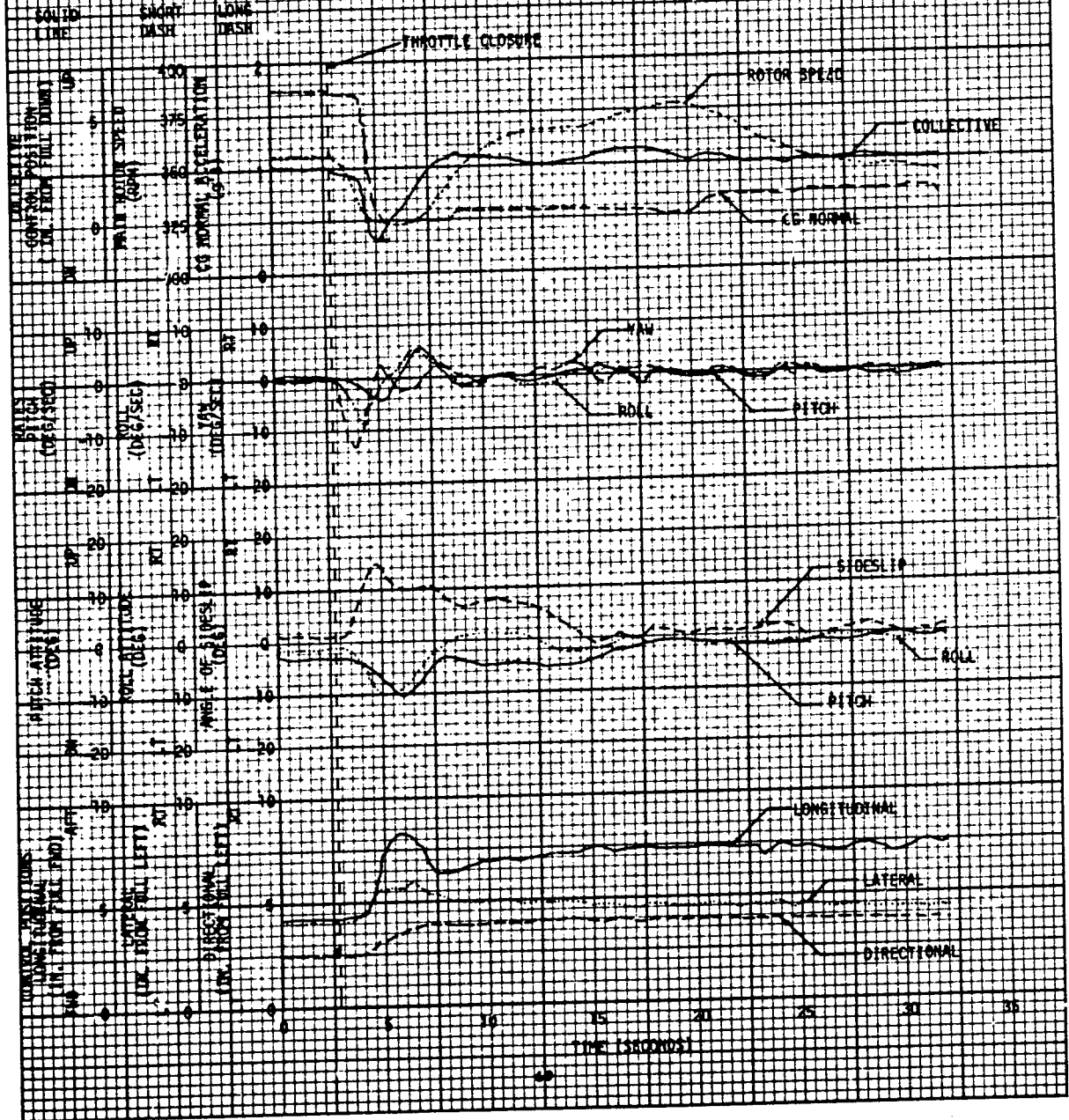
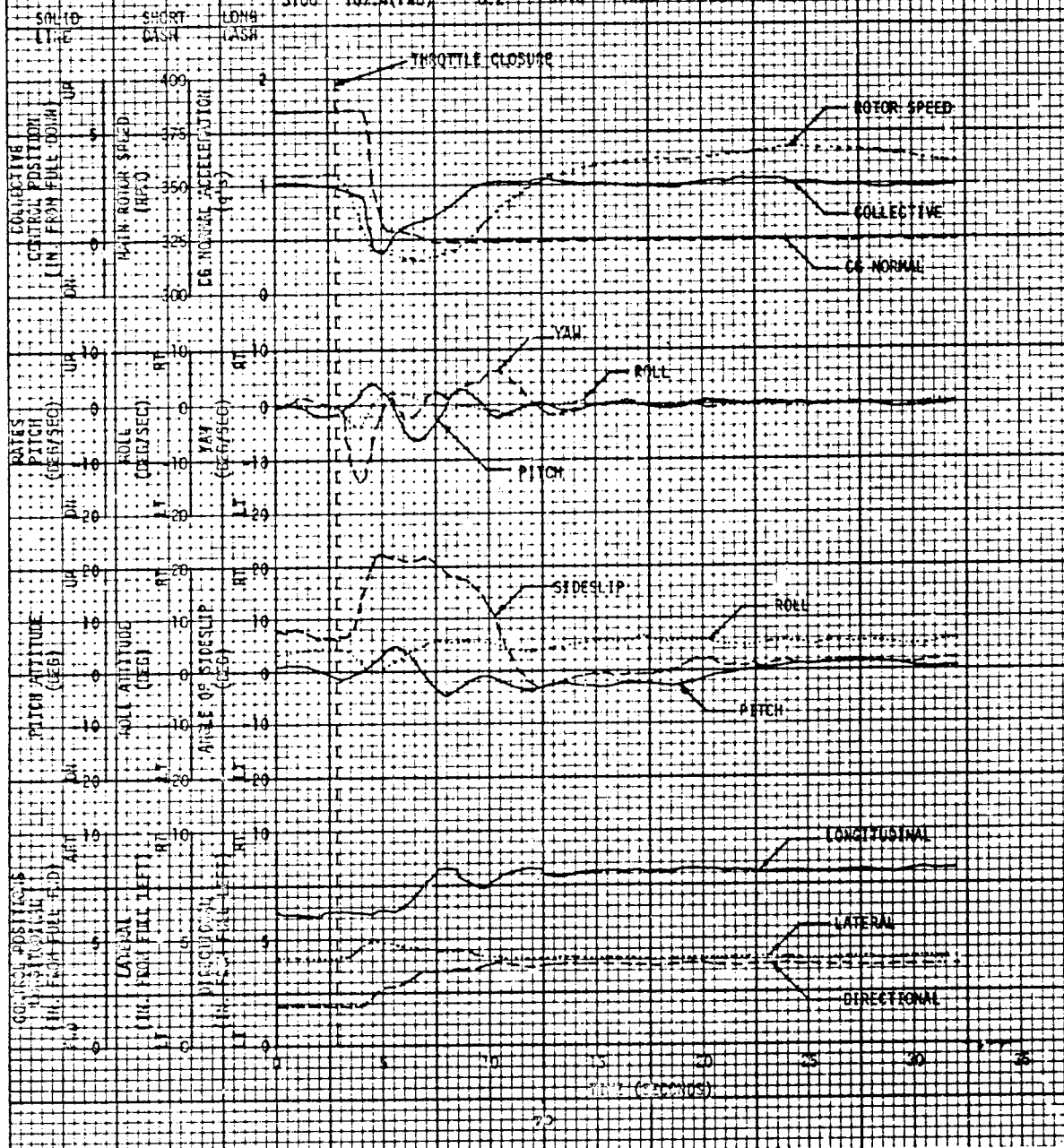


FIGURE 26

SIMULATED SUDDEN ENGINE FAILURE
CH-53C USA S/N 68-16350

GROSS WEIGHT (LB)	CG LOCATION (FWD)	DENSITY (BL)	TRIM ALTITUDE (FT)	TRIM AIRSPEED (KIAS)	TRIM FLIGHT CONDITION
3100	107.3 (FWD)	0.2	5040	115.5	CLIMB



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